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THE ORBITING GEOPHYSICAL OBSERVATORIES

by George H. Ludwig

Goddard Space Flight Center
Greenbelt, Md.

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SUMMARY

The Orbiting Geophysical Observatories and the supporting ground checkout equipment, data acquisition and tracking stations, and data processing equipment are designed to conduct a large number of diverse experiments in space. Measurements will be made within the earth's atmosphere, ionosphere, exosphere, and magnetosphere, and in cislunar space, to provide a better understanding of earth-sun relationships and of the earth as a planet. The observatories include six booms to support detectors away from disturbances generated in the main body of the spacecraft. Five degrees of freedom allow the orientation of experiments relative to three references–the earth, the sun, and the orbital plane. Power, thermal control, and data handling subsystems provide for the proper operation of the experiments and the telemetering of the data. Ground stations receive the data which are then processed into a form suitable for use by the experimenters. The systems have been designed to make available a standard type of spacecraft and support equipment which can be used repeatedly in launching a large number of easily integrated experiments in a wide variety of orbits.

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INTRODUCTION

During the five years since the launching of the first artificial earth satellite, satellite systems have undergone a rapid evolution. For investigations in the space sciences, two principal satellite types are now in use. The first is the relatively small satellite which includes the Explorer series, the Vanguard series, the State University of Iowa Injun series, the Naval Research Laboratory Lofti and Solar Radiation series, and the international satellites Ariel and Alouette. In general, these spacecraft contain sets of directly related experiments, provided by a small number of laboratories or, in a few cases, a single laboratory. The second type is the comparatively large orbiting observatory. Three observatory series are now in use or planned, the Orbiting Solar Observatory (OSO), the Orbiting Geophysical Observatory (OGO) described in this paper, and the Orbiting Astronomical Observatory (OAO). These spacecraft are designed to carry large numbers of easily installed experiments of a somewhat more diverse nature than those carried in the small satellites.

The Explorer Type Satellites

Because of launch vehicle limitations and the need for reliability in a new technology, the first satellites (i.e., the Explorer type) were small, light, and simple. As larger vehicles such as Delta became available, the weight and size of these spacecraft increased. And as the technology advanced, spacecraft complexity tended to increase to provide more information from each launching. We may, however, expect continued use of the Explorer type satellite, since it has several advantages over the larger, more complex observatories. These may be summarized as follows:

1. Some experiments may require orbits into which the observatories cannot be placed because of launch vehicle limitations; or the number of experiments requiring a particular orbit may not warrant the use of an observatory.
2. The experiments may require a different spatial orientation than is contemplated for any of the observatories. For example, a spin-stabilized satellite may be more suitable

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for certain experiments which need to rapidly scan a large range of directions on the celestial sphere. Magnetic field orientation may be preferred for certain groups of charged-particle experiments.

3. Some classes of very sensitive experiments may require small satellites to avoid contamination or interference from the satellite structure, power and electronic subsystems, or other experiments.
4. It is possible to launch a small satellite with a very short lead time for the high priority investigation of new phenomena. At the present time this is feasible only under exceptional circumstances. An example is the launching of Explorer XV (1962 $\beta\lambda 1$) for the investigation of the high flux of the geomagnetically trapped particles injected by the Starfish high altitude nuclear explosion in the summer of 1962.
5. During the early phases of the observatory program the reliability of the small satellites may be higher than that of the observatories. This advantage may disappear later since the larger weight capability of the observatories will permit a higher degree of redundancy, and because the repeated use of the standard observatory designs should lead to a continuous increase in reliability. This point is speculative at this time, since only one observatory OSO I (1962 ζ 1) has been launched thus far. Interestingly, its operating lifetime has been longer than the average lifetime of the recent Explorer type satellites.
6. A number of experimenters feel that the small satellite with its simple telemetry system and spin or magnetic field stabilization offers a more nearly ideal tool for research and university student training. They feel that they are able to retain more direct control over their investigations. The organizational structure is simpler for the smaller satellite programs, and less effort is required for liaison with other groups and for the planning of the operational aspects such as prelaunch testing and data processing. This may not necessarily continue to be true.

The Observatories

The small satellites have tended to be highly integrated mechanically, thermally, and electrically in order to take full advantage of the launch vehicle capability; i.e., they are built as tightly knit, homogeneous assemblies. Thus, it was necessary to almost completely disassemble the Explorer I (1958 α1) and Pioneer IV (1959 ν1) spacecraft in changing the batteries. This situation has steadily improved with the availability of the Delta launch vehicle, and the internal satellite systems have tended to become separated into more easily changeable subassemblies. But a moderately high degree of electronic system integration is still employed. This means that these spacecraft will continue to be essentially one-mission systems in the sense that extensive redesign of the subsystems will be necessary to accommodate each new set of experiments.

As the larger launching vehicles such as the Thor-Agena and Atlas-Agena have become available, it has been possible to launch heavier experiments and larger numbers of small experiments. In addition, it has become possible to increase the capabilities of the data handling, power, and thermal subsystems to allow greater flexibility. And solar, earth, and inertial reference attitude control systems have become feasible. It has also become possible to establish well-defined, standard electrical and mechanical interfaces between the experiments and the various spacecraft subsystems. Thus, the concept of a family of standard observatories has evolved. These spacecraft are standard in the sense that they present a well-defined set of interfaces to the experiments and are highly flexible in order to accommodate many types of experiments. It should not be necessary to design and develop a new spacecraft for each mission; a spacecraft design can be used repeatedly, with only minor modifications, for different combinations of experiments. The advantages of the standard observatory include:

1. Large numbers of directly and indirectly related experiments can be performed concurrently to study the correlations between several phenomena at given positions in space. For example, with the OGO it will be possible to study simultaneously the relationships between solar events, the solar plasma, the earth's trapped radiation belts, the earth's magnetic field, and the atmospheric structure.

2. A few heavy or bulky instruments can be launched to perform more extensive, complex, or detailed observations. The OGO will be able to carry a large earth-oriented experiment weighing more than 68 kg.

3. The electrical, mechanical, and thermal interfaces between the experiments and spacecraft subsystems are well defined and will remain essentially fixed from mission to mission. It should be possible to avoid difficulties encountered in previous projects when the experiments and spacecraft were developed concurrently without the benefit of previously existing definitions of the interfaces between them. The subsystems have enough flexibility that they should not seriously limit the evolution of experiment technology for some time.

4. The system reliability should be improved ultimately by the repeated use and continuous improvement of the basic design, and by the fact that the larger weight allotment will permit a higher degree of redundancy.

5. The continued use of a standard spacecraft design should lead to higher operating efficiency through the continuous evolution of a ground data acquisition and tracking station network, data processing equipment, and operating procedures. The use of the larger spacecraft will reduce the total operational load, since a given number of experiments will be carried on a smaller number of spacecraft.

6. In spite of the rather high development cost of the observatories, the ultimate cost of orbiting a given weight of experiments should be lower than if they were carried on a greater number of small satellites, since the development of a new spacecraft for each new mission will be avoided.

THE OGO PROJECT

The Orbiting Geophysical Observatory project includes the development and use of the experiments and spacecraft which make up the observatory, the ground checkout equipment, the network of ground receiving and tracking stations, and the data processing equipment. It is a part of the NASA long range program in space sciences.

Once the experiments for each launching are selected by NASA Headquarters, contracts are written by Goddard for the financial support of the experimenter's efforts. The experimenters are furnished with the necessary technical information to enable them to design their instruments, plan their integration and testing, and plan their data processing after launch. The experimenters have the responsibility of delivering finished experiments which will meet their initial objectives and survive the environmental tests. The experimenters also have the primary responsibility for processing and analyzing the data obtained from their experiments. They will receive digital computer tapes containing the raw data from their own experiments in cleaned and sorted but otherwise unprocessed form, and the necessary housekeeping and timing information. A different set of tapes will contain the orbit and observatory orientation information. The experimenters will arrange for computer programming, processing, tabulation, analysis, etc., in keeping with the belief that the typical investigator prefers to retain direct control over this phase of the operation.

Orbits and experiments have been chosen for the first five OGO missions. The first, third, and fifth OGO's, the Eccentric Orbiting Geophysical Observatories (EGO's) will be launched by Atlas Agena B's in 1964, 1965, and 1966 from the Eastern Test Range into highly eccentric orbits having initial perigee and apogee heights of 280 and 110,000 km above the earth, respectively, and initial orbital inclinations of approximately 31 degrees. The second and forth OGO's, the Polar Orbiting Geophysical Observatories (POGO's) will be launched into near-polar orbits in 1964, 1965, and 1966. Thor Agena launch vehicles will be used at the Western Test Range. The POGO orbits will have perigee and apogee heights of 250 and 920 km above the earth.

DESCRIPTION OF THE SPACECRAFT

The OGO may be considered to consist of two parts, the experiments and the spacecraft. The spacecraft was designed by the TRW/Space Technology Laboratories, Redondo Beach, Calif. It consists of a basic structure to support and enclose the experiments and other assemblies, an attitude stabilization subsystem, and power, data handling, communications, and thermal control subsystems for servicing the experiments. The weight of the spacecraft is approximately 399 kg, and it is designed to accommodate about 89 kg of experiments, making a total observatory weight of about 488 kg.

Configuration

The configuration of the deployed observatory is shown in Figure 1. The central box structure measures 1.70 m long by 0.78 m high by 0.81 m wide. Its size was chosen to provide a large internal volume which would accommodate an assortment of large irregularly shaped experiments in addition to the spacecraft subsystem components. It was limited, however, by the requirement

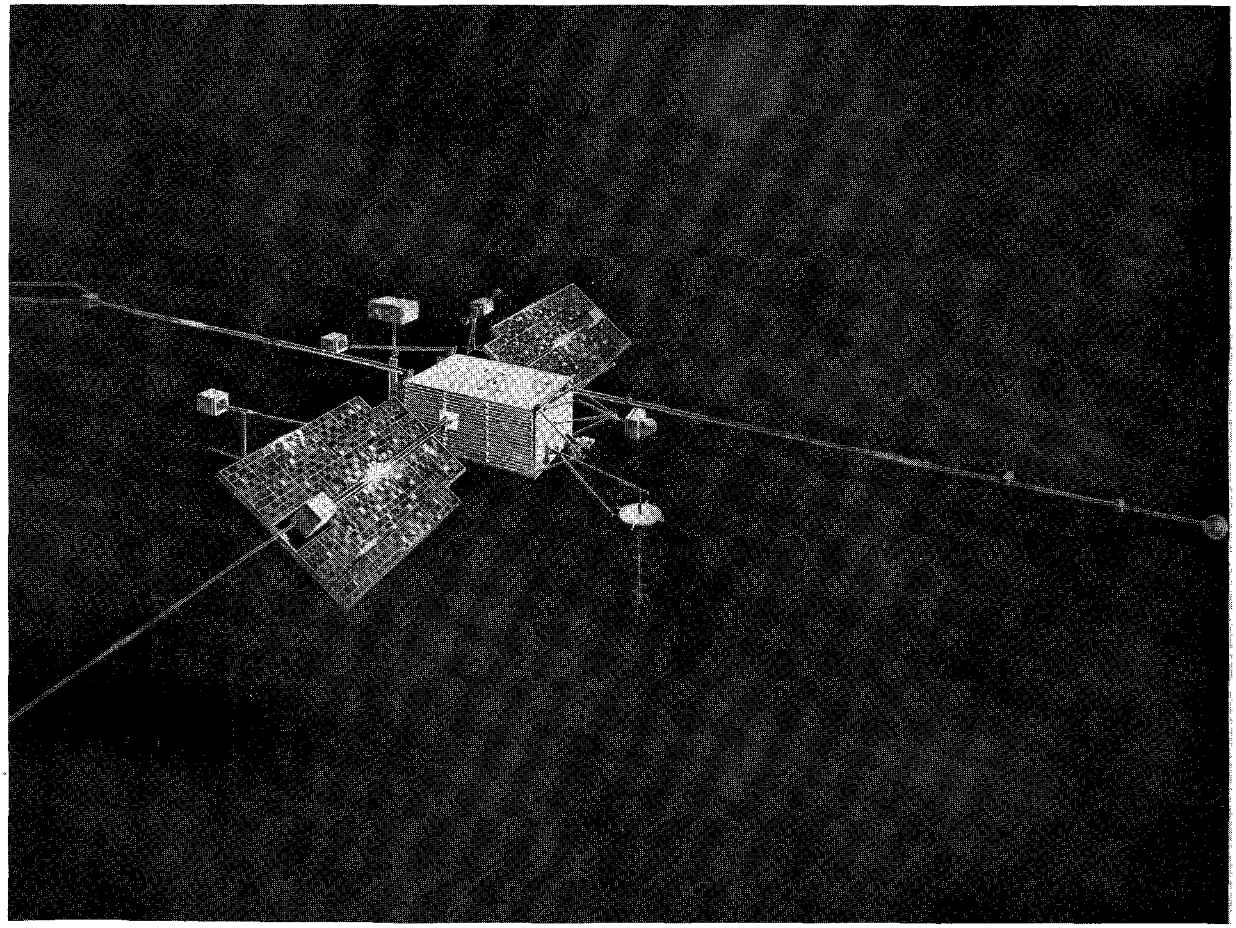

Figure 1—The Orbiting Geophysical Observatory. The appendages show the experiments for the first flight.

that it fit within the standard nose fairing for the Agena rocket. The action of the attitude control system causes one of the 0.81 by 1.70 m faces of the main body to face the earth at all times. Experiments which require earth orientation are mounted on this face, and those requiring orientation away from the earth are located on the opposite side.

The solar panels which provide the electrical power for the observatory are mounted on a shaft passing through the main body. The attitude control system orients the silicon solar cells, mounted on these panels, toward the sun, by controlling the rotation of the main body about the earth-observatory axis and by controlling the rotation of the solar panels about their shaft axis. Enclosures located on each solar panel contain experiments requiring a fixed orientation with respect to the sun. The orientation of the panels toward the sun results in the orientation, at right angles to the observatory-sun line, of the two main body faces through which the shaft passes. Thus these two faces are never illuminated by the sun and are used to radiate excess heat from the observatory.

A pair of experiment containers is located on another shaft which rotates about the observatory-earth axis. This rotation is controlled by the attitude control system so that the detector axes fall in the orbital plane. Thus, the angle between the detector axes and the observatory velocity vector will be simply related to the true anomaly. This angle will always be zero at apogee and perigee; it will also be zero throughout the entire orbit if the orbit is circular. Experiments designed to investigate the characteristics of particles whose velocities are not large compared with the velocity of the spacecraft will be located in these containers.

Booms extending from the ends of the main body support experiments at some distance from the central subsystem assemblies. They are intended for detectors whose measurements might be affected by disturbances generated in the main body and in the solar panels. For example, the magnetometer sensors are located at the ends of long booms so that the magnetic fields at the sensors produced by ferromagnetic materials and electric currents in the spacecraft are smaller than the interplanetary magnetic fields being investigated. Isolation is also necessary for investigations of portions of the electromagnetic wave spectrum, experiments sensitive to the outgassing from the main body, and experiments which cannot tolerate the proximity of appreciable mass. Two long booms approximately 6.6 and 6.2 m in length and four short booms, each approximately 1.3 m in length, are included, as indicated in Figure 1. The distance from the tip of the near boom to the tip of the loop on the far boom is approximately 17.7 m. The distance between the ends of the solar panels is about 6.0 m.

Some of the antennas for the communication system are also supported on these booms. And the high gain antenna for the wideband digital data transmitter is supported by an additional boom so that it will not obstruct the view of experiments mounted in the main body. Two more booms provide large moments for the cold gas jets which apply torques to the observatory main body to help control its orientation.

The observatory is designed to fold into a launch configuration which will fit within the 1.65 m outside diameter of the nose fairing. The structural design model of the folded OGO is shown in Figure 2. The various appendages can be seen folded against the main body in their launch positions. After injection into orbit, pneumatically actuated latches release all the appendages, and they are driven to the open positions by spiral springs located in the hinge joints. Levers and detents rigidly lock the joints in an open position.

Structure

The basic structure of the main body can be seen in Figure 3. The panels which form the sides are made of lightweight corrugated aluminum sandwich sheets to give the required stiffness, thermal conductivity, and ease in attaching assemblies. Four longerons in the corners of the spacecraft together with the four vertical side panels absorb the acceleration loads during launch and transmit them to the four supports at the bottom of the main body. The loads are carried from these four legs to the upper ring of the Agena by the four inverted vees of the interstage structure (Figure 2). The observatory is held to the Agena during launch by a tension band with four shoes which clamp the four supporting feet to the interstage structure. Upon the reception of the

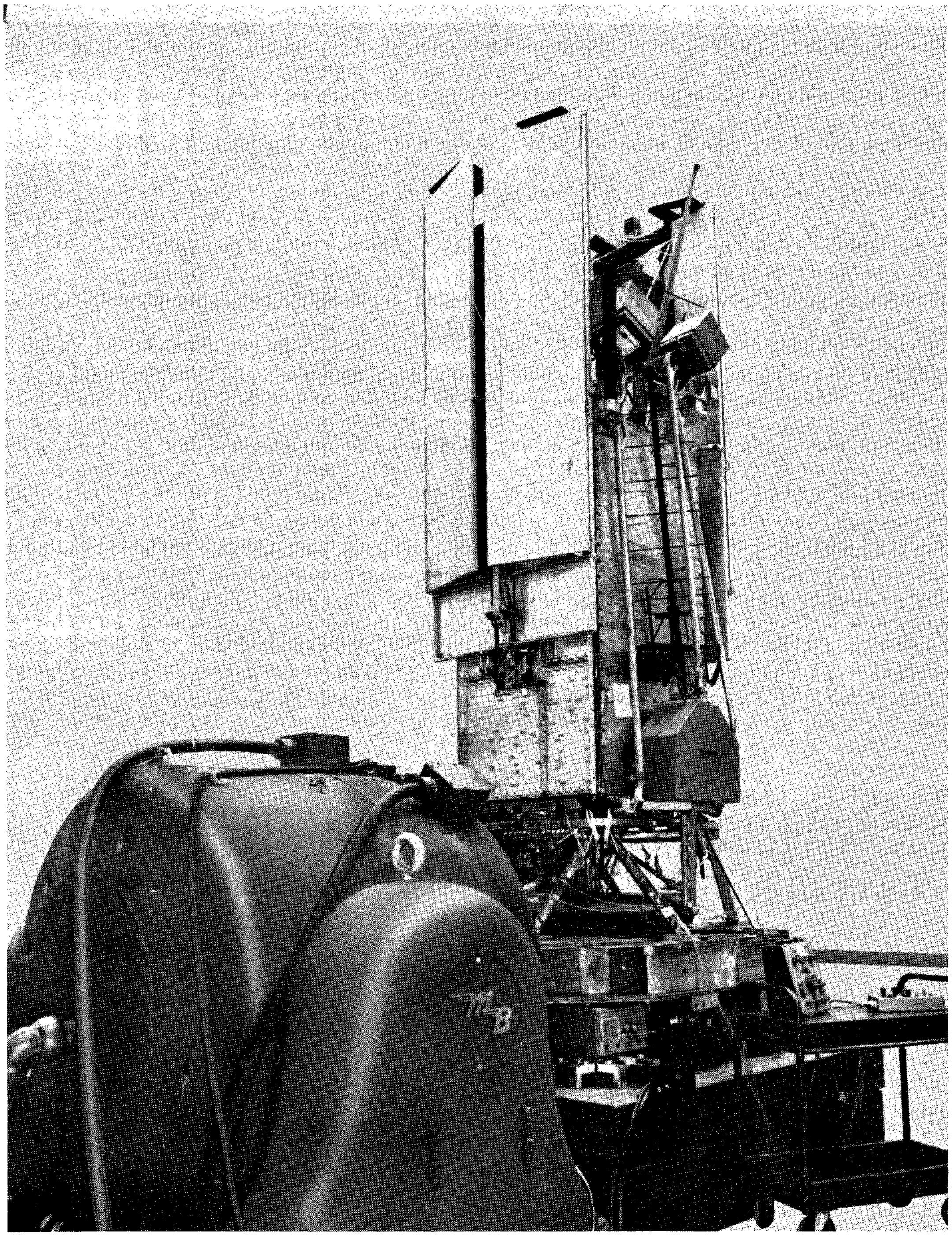

Figure 2—The observatory in its folded configuration. The structural design model with generalized appendage experiment containers is mounted on the vibration table for one of the two transverse vibration tests. Only a few of the thermal control louvers are mounted and none of the solar cells are in place. The main body is mounted on the adapter by which it will be attached to the Agena rocket. (Photograph courtesy of the Space Technology Laboratories, Inc.).

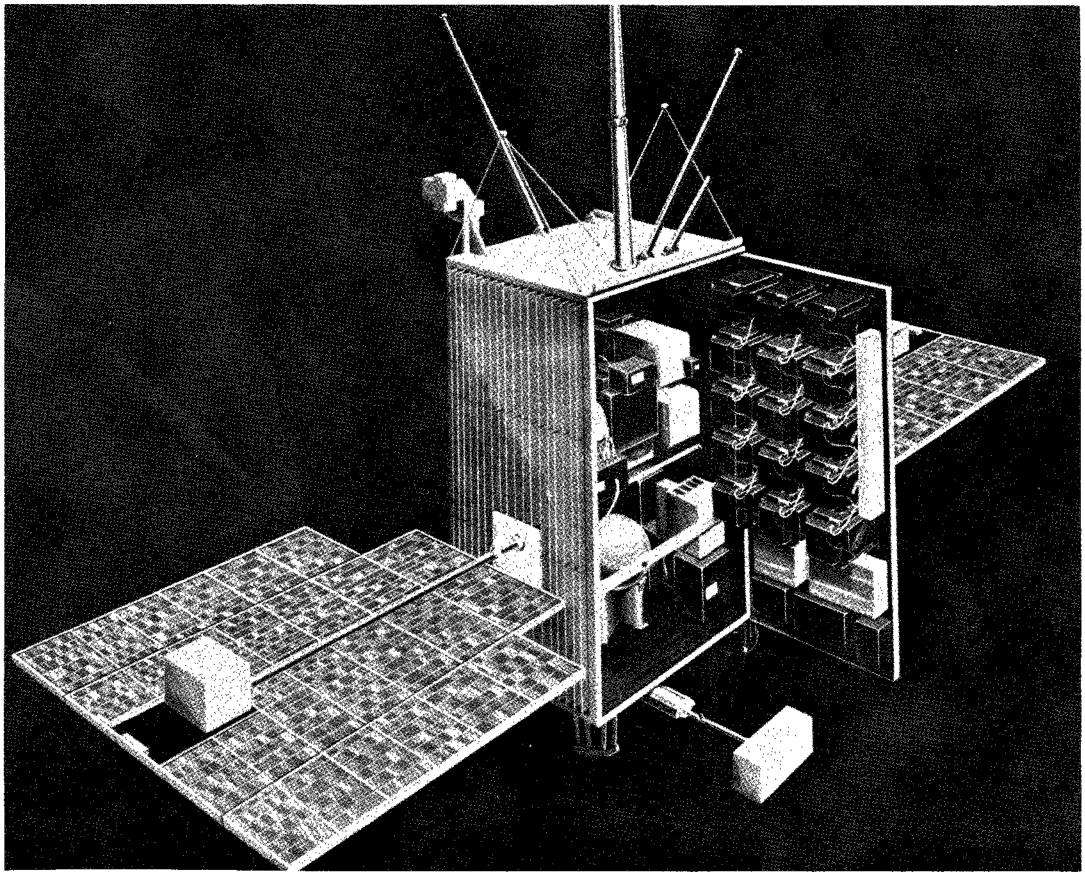

Figure 3—The main body with one of its two doors opened. Experiments are mounted on the upper two-thirds of this door and the corresponding door on the opposite side. Subassemblies for the servicing subsystems are mounted inside the main body. (Many appendages are not shown.)

separation signal from the Agena following injection, explosive actuators release the tension band, and coil springs located at each of the four feet impart a separation velocity of approximately 1.5 m per sec. The spring tensions are carefully matched so that the angular velocity imparted to the observatory upon separating is less than 1 degree per second.

The two sides of the main body through which the solar array shaft passes are designed to serve as efficient heat radiators. These panels are covered with a grid of louvers whose positions are controlled thermostatically to vary the exposure of the radiating surfaces. The other two side panels are hinged in sections to provide easy access to the interior. These doors are securely closed in flight by fasteners around their circumferences. Additional rigidity is given to the main body structure by removable internal braces, the solar array shaft assembly, and an intercostal structure attached to the thermal radiating panels.

The spacecraft subsystem internal assemblies are attached to the two thermal radiating panels, the intercostal structure, and the lower door sections. A large volume inside the main body is reserved for experiments. The shape and size of this region provides great flexibility for accommodating a large variety of experiment configurations. One possible arrangement is shown in Figure 3. For convenience in planning the experiment locations, the upper two-thirds of each door is divided into 15 basic modular areas, each measuring 20.32 cm square. The lower nine squares on each door are clear to depths of 20.32 cm behind the door. The upper six squares are clear all the way through the body. Each experimenter having an assembly in the main body is assigned an integral number of these squares. He may use a portion of each modular area or the entire area, or he can utilize several squares for a single assembly. Since the upper portion of the main body is completely unobstructed, large experiments can be located in that region. A single experiment as large as 61.0 by 40.6 by 76.2 cm can be accommodated.

The OGO structure can support 194 kg in addition to the 488 kg of the basic spacecraft. This growth potential will be used when larger launch vehicles become available to accommodate a larger experiment load. Included in the growth potential is the capability of carrying and separating in orbit a 135 kg auxiliary satellite to perform experiments requiring an especially undisturbed environment or to perform mother-daughter experiments where the large separation between this small satellite and the main observatory will be useful.

Each solar array panel consists of a lightweight aluminum framework to which the solar cell modules are attached. Figure 4 shows one of the panels with its solar cell module plates attached. The 144 module plates can be removed independently; this simplifies the replacement of any cells which may fail during the observatory testing program. The panel is fastened to the solar array shaft by a hinge member at the right of the figure.

The orbital plane experiment containers are supported by a trussed structure at one end of the main body. These containers, their supporting shafts, and their cylindrical drive assembly can be seen on the lower end of the observatory in Figure 2.

The booms are made of lightweight aluminum tubing. They are hinged at the main body end in order that they can be folded for launch. The two long booms each consist of three short sections which fold against the main body in the launch configuration. Electrical cables to the experiments in the containers are routed through the booms. The fittings for attaching the containers to the ends of the booms and those for attaching the booms to the main body are made of thermally and electrically insulating epoxy impregnated glass fiber. The thermal insulation is necessary to prevent heat loss through the booms from the temperature-controlled boom containers and the main body. The electrical insulation permits control of the boom potentials by experiments designed to study charged particles having thermal energies.

Standard mounting fittings permit the easy attachment of the solar, orbital plane, and boom experiment containers. These fittings define the mechanical interfaces between the experiments and the spacecraft. The appendage container designs are tailored to the needs of the experiments in order that the somewhat-limited weight capabilities at these positions may be utilized most efficiently. The structure is designed to support 7.17 kg at each solar experiment container attachment point, 7.35 kg at each orbital plane experiment container attachment point, and 3.27 kg at each boom experiment attachment point.

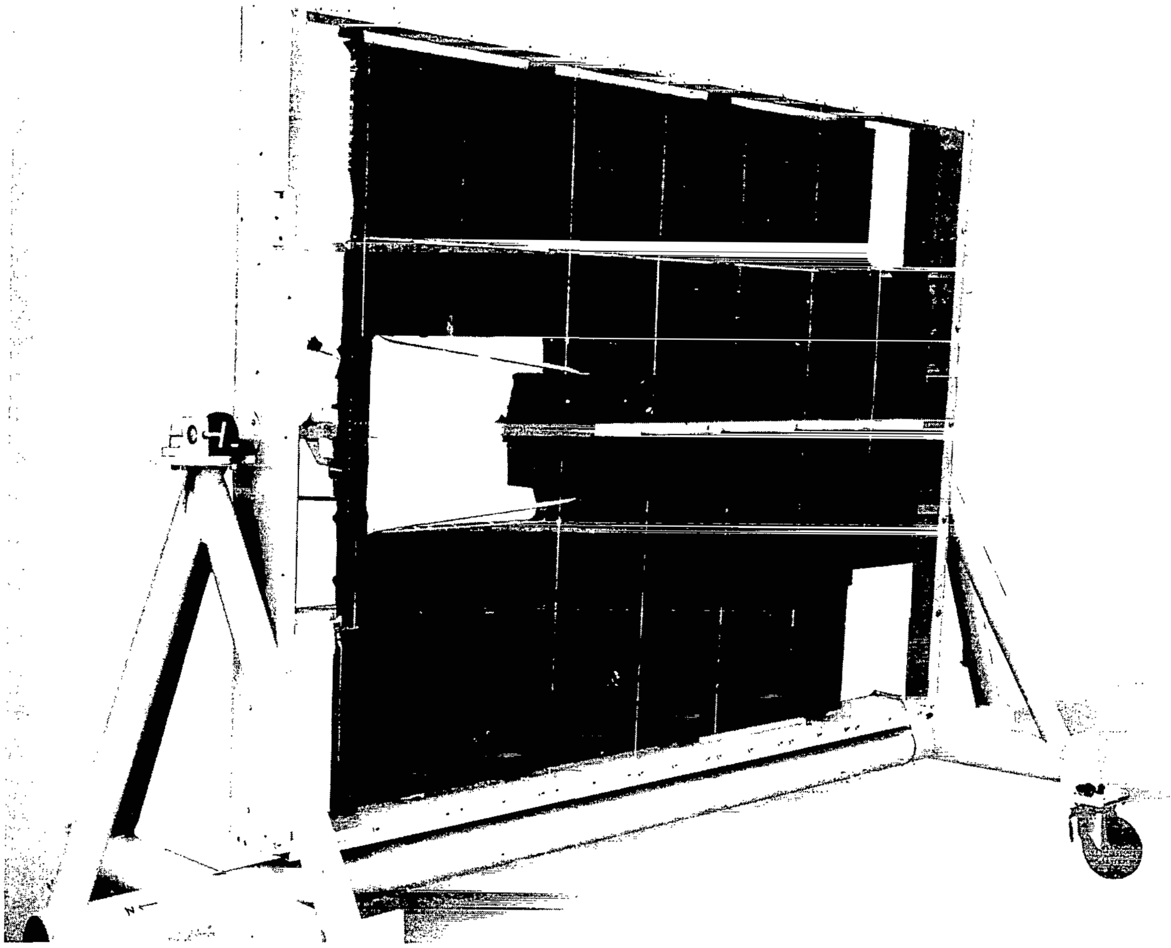

Figure 4—One of the two solar array panels. Each of the 144 module plates contain 112 gridded p-n junction silicon cells. (Photograph courtesy of the Space Technology Laboratories, Inc.)

For the attitude control system to operate properly, it is necessary that the principal axes of the complete observatory be properly positioned. This is done by careful distribution of the experiment mass in the observatory and by adjustment of the long boom positions. Swivel joints and adjusting screws at the bases of these booms permit adjustment over a range of approximately 10 degrees in two planes.

The release of gases from the observatory is controlled to minimize interference to experiments studying the characteristics of the atmosphere. The attitude control system gas jets are directed away from all experiment enclosures. Small volumes such as the tubular booms are freely vented to prevent the transport of gases from one position in the orbit to another. Large volumes such as the main body and appendage experiment containers are vented away from all experiment mounting positions. Special care is taken to avoid venting gas toward the orbital plane experiment containers, since the atmospheric structure experiments are usually located there.

Thermal Control

The thermal control subsystem has been designed to maintain the temperatures of all the assemblies in the main body of the spacecraft within the limits of 5° and 35°C. Since the orientation of the main body with respect to the sun is variable, and since the satellite may spend periods as long as 2 hours in the earth's shadow, it is necessary to use an active thermal control system. The use of an active system also makes it easier to accommodate large variations in experiment power dissipation and in the sizes of sensor openings through the external surfaces.

Thermal input to the spacecraft from the sun and earth is reduced to a very low value by an efficient radiation shield, and the thermal radiation from the body is controlled by variable-area radiation panels. The radiation shield, consisting of multiple layers of aluminized Mylar, covers all areas of the main body which may be exposed to the sun. The two sides of the main body through which the solar panel shaft passes and the end of the main body at which the orbital plane experiments are located are never exposed to the sun, because of the action of the attitude control system. These three surfaces are efficient heat radiators and are covered with thermal insulation louvers to control their exposure. Each louver is positioned by a bimetallic spring which senses the temperature of the radiating panel. When the temperature of the radiating panels rise, the louvers open to allow the radiation of more heat. The construction of one of the thermal control panels can be seen in Figure 5, which shows the radiating surface, the louvers, and the bimetallic elements. The efficiency of this thermal control system is demonstrated by the fact that calculations indicate that the temperature of the main body will not lower significantly as long as the normal operating electrical power is dissipated.

Thermal control of the appendage experiment containers is obtained by using thermal radiation shields, radiation surfaces, and electrical heating. The containers are thermally insulated from their mountings. The radiation area sizes and locations are chosen to provide a proper heat balance during periods of maximum energy input. Electrical heaters in the containers supply additional energy during long eclipses or when the experiment power is turned off. With this system the temperatures of the experiment assemblies within the appendage containers normally will be between 0° and 40°C.

Experiment sensors that protrude through the radiation barriers on either the main body or the appendage containers present special thermal problems. They must be designed so that the solar energy flux, about 1400 w/m^2, does not cause excessive heating of the sensors, and so that the thermal radiation when the sensors are not illuminated by the sun does not cause excessive cooling. In some cases it is necessary to allow greater temperature excursions than those quoted above for sensors having large openings.

The solar array presents a particularly difficult thermal design problem. Less than 10 percent of the 10,000 watts of solar energy incident on the two solar panels is converted into electrical energy. Most of the rest is radiated from the rear surfaces of the panels. A potassium silicate compound having a high infrared emissivity and a low visual and UV absorptivity is used on these back surfaces to keep the solar cell temperatures below approximately 85°C when the

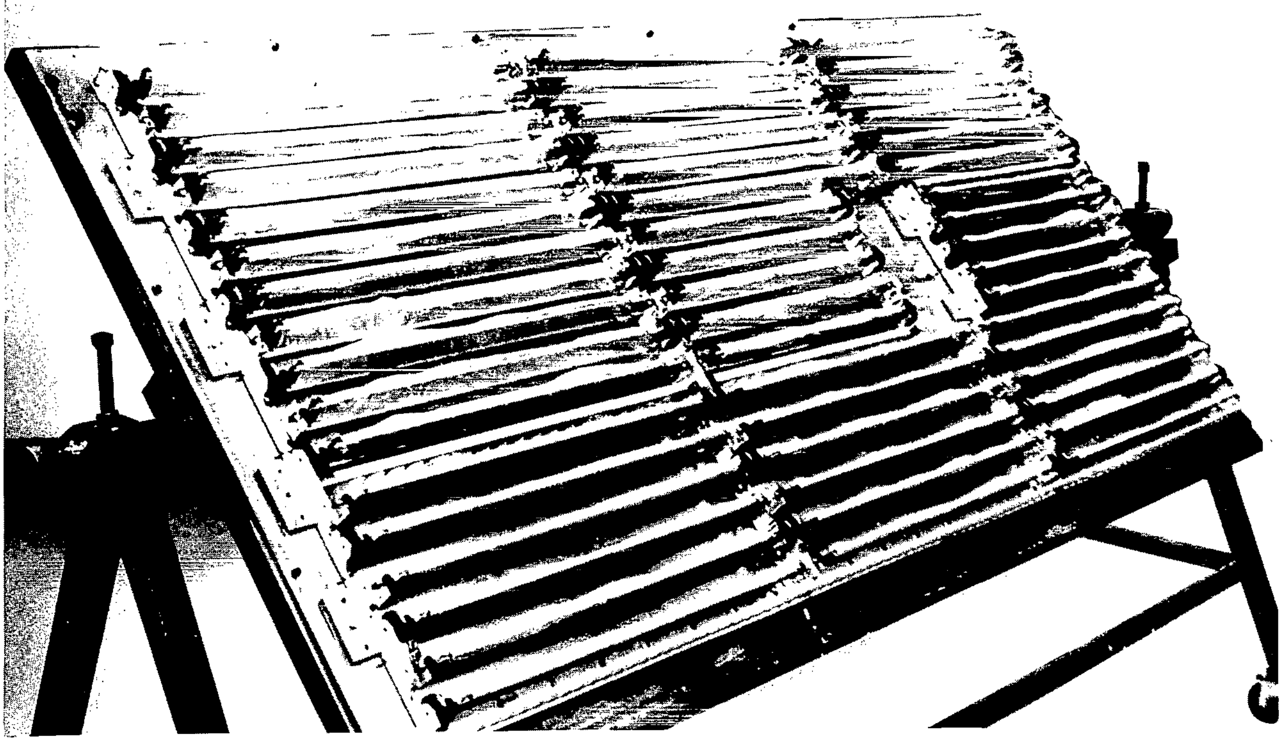

Figure 5—One of the two louvered thermal control panels. The louvers are individually controlled by bimetallic thermostats which sense the temperature of the high emissivity radiation panel underneath. (Photograph courtesy of the Space Technology Laboratories, Inc.)

panels are fully illuminated. It is necessary to keep the solar cell temperatures above -140°C during eclipse to avoid damage to the cells by thermal stresses. This is accomplished by making the beryllium solar cell substrates thick enough to provide sufficient thermal mass to prevent cooling below -140°C after 2 hours in darkness.

The solar cell energy spectral response peaks in the blue region. The red region of the solar spectrum, which contributes thermal energy without contributing to the electric power generation, is rejected by an electrodeposited optical coating applied to 0.15 mm thick glass slides which cover the solar cells. These glass slides also provide a limited measure of protection from charged particle radiation and micrometeoroid damage.

Power Supply

The power supply subsystem consists of a solar energy converter, a chemical battery, a charge regulator, and power distribution equipment. The solar energy converter consists of 32,256 gridded p-n junction silicon cells, each having an effective area of 1.9 cm^2 and a solar energy conversion efficiency in space of 10.5 percent. The individual cells are mounted in groups of 112 on beryllium plates or substrates, as shown in Figure 4. Beryllium is used because of its light weight and because its coefficient of thermal expansion is similar to that of the silicon cells. The cells on the 7 by 16 cell module are wired in series-parallel. One hundred and forty-four of these modules are attached to each of the two solar panels. The complete array can supply an initial total electrical power of approximately 650 watts. Allowance for losses due to light transmission through the glass cover slides, orientation errors, and errors in measurement and cell matching gives an initial effective available power of approximately 490 watts at 29.5 volts. Degradation during the 1 year operating period due to damage from charged particle radiation and micrometeoroid bombardment may reduce the power output to approximately 300 watts. Of this, at least 50 watts is reserved for operation of the experiments. The rest is used by the attitude control, thermal control, data handling, and communications systems.

Two nickel-cadmium battery packs, each of 12 ampere-hours capacity, provide electrical power to the observatory during eclipse and assist in gross regulation of the power bus voltage. Each battery pack consists of 22 prismatic cells and weighs about 15 kg. They are mounted directly on the two radiating faces of the main body to prevent excessive battery temperatures. Half of each battery pack is electrically connected in series with half of the other pack, to equalize the heat dissipated from the two packs between the two radiating panels. The battery size has been selected so that the depth of discharge during the 2 hour EGO eclipse will be limited to 75 percent. When the spacecraft is placed in a near-earth orbit, with more frequent, 35 minute eclipses, the discharge depth will be limited to 25 percent. A silver-cadmium battery is also being developed for use on later observatories. This battery is nonmagnetic and therefore more compatible with the magnetic field experiments. It is expected to have a somewhat higher efficiency in terms of stored energy per unit mass.

Charge control equipment regulates the charging of the chemical batteries by the solar arrays. Two regulators maintain a preset charge current to the two batteries by shunting the unneeded portion of the array output current through power transistors. These transistors are mounted on heat sinks at the outboard ends of the arrays. One of several preset charge current rates can be chosen by ground command to fit the sunlight-eclipse ratio and power load conditions. Sensors reduce the charge rate to a trickle charge when the chemical battery is fully charged or its temperature exceeds 35°C. If the battery temperature exceeds 52°C, because of battery failure, a relay transfers operation to the remaining battery. Should both batteries fail, the solar array will furnish power directly to the electrical power bus.

The batteries are connected directly to the primary 28 volt bus which provides power for the attitude control, command receiver, and data handling subsystems. A secondary bus, connected to the primary bus through an undervoltage cutoff relay, provides power to the experiments and the rest of the communications system. The undervoltage cutoff relay disconnects the secondary bus whenever the voltage falls below 23.5 volts. This is not expected to occur unless there is excessive damage to the solar array, orientation is lost, or there is an excessive power drain. The undervoltage relay is reset by ground command.

The power bus upper voltage limit is set at 33.5 volts by the charge control equipment. Therefore, experiments must be capable of operating over the range of input voltage from 23.5 to 33.5 volts. Experiments and spacecraft systems which require other voltages or better regulation employ converters and regulators. The converters employing chopping circuits for their operation are synchronized at frequencies of 2461 or 7384 cps to minimize interference to experiments designed to investigate the VLF portion of the electromagnetic wave spectrum. Motors in the attitude control system are operated from a 400 cps power inverter.

Attitude Control

The OGO has five degrees of freedom: rotation of the main body about each of its three principal axes, rotation of the solar array with respect to the main body, and rotation of the orbital plane experiments with respect to the main body. The attitude control system controls those motions to meet the orientation requirements of the experiments, solar array, thermal radiating surfaces, and directional antenna. The rotations of the main body about the longitudinal (pitch) axis and about the solar array (roll) axis are controlled so that one of the main body experiment mounting surfaces and the directional antenna are directed toward the center of the earth with an accuracy of ±2 degrees. The rotation of the main body about the satellite-earth (yaw) axis and the rotation of the solar array about its shaft axis are controlled so that the solar array and the solar experiments are directed toward the sun, and the two main-body thermal-control side panels are aligned perpendicular to the sun line. The sun pointing accuracies are normally ±5 degrees. The rotation of the orbital plane experiments with respect to the main body about their mounting axis (parallel to the body yaw axis) is controlled so that these experiments are always directed in the orbital plane with an accuracy of ±5 degrees whenever the main body angular rate about an axis normal to the orbital plane is greater than about 1.1 degrees per minute. Near the apogee of a highly eccentric orbit this angular rate is lower, and the orbital plane orientation error may become much larger.

The operation of the attitude control system can be seen in Figure 6. This diagram shows the system in its normal mode of operation. The earth horizon is detected by infrared edge-sensing scanners. Horizon scanner logic circuits determine the earth's center over the entire range of satellite-earth distance from 280 km to greater than 17 earth radii, where the angle subtended by the earth ranges from about 150 degrees to about 6 degrees. The sun subtends too small an angle to be accepted as a tracking source. The error signals produced by the scanner logic are amplified and applied to motors which drive the roll and pitch inertia wheels. The reactions to the angular acceleration of these wheels produce torques which rotate the observatory to reduce the earth orientation errors. The reaction wheel servo systems have central dead-bands, so that no power is applied to the drive motors until the errors exceed 0.4 degree. These same error signals are applied in parallel to argon gas jets, with somewhat wider dead-bands. Thus the cold gas jets are used only when large errors occur due to the nonsymmetric build-up of main body external torques from unbalanced aerodynamic forces, magnetic field interactions, solar radiation pressure, and the gravity gradient. The torques needed to satisfy the orientation requirements are very nearly periodic over an orbit, and can be supplied by the reaction wheels. Thus, the gas jets are not expected to operate more than about once per orbit. This low duty cycle for the gas jets and the use of argon gas rather than a gas having a lower atomic weight are necessary to avoid interference with the experiments which measure the atmospheric composition.

The direction to the sun is sensed by silicon p-n junction cells similar, in general, to the solar cells used for power generation. They are used in pairs, with baffles between the two cells of each pair. The difference in the outputs from the cells in each pair are a measure of the error. Coarse and fine sensors are included, for use during sun acquisition and normal operation

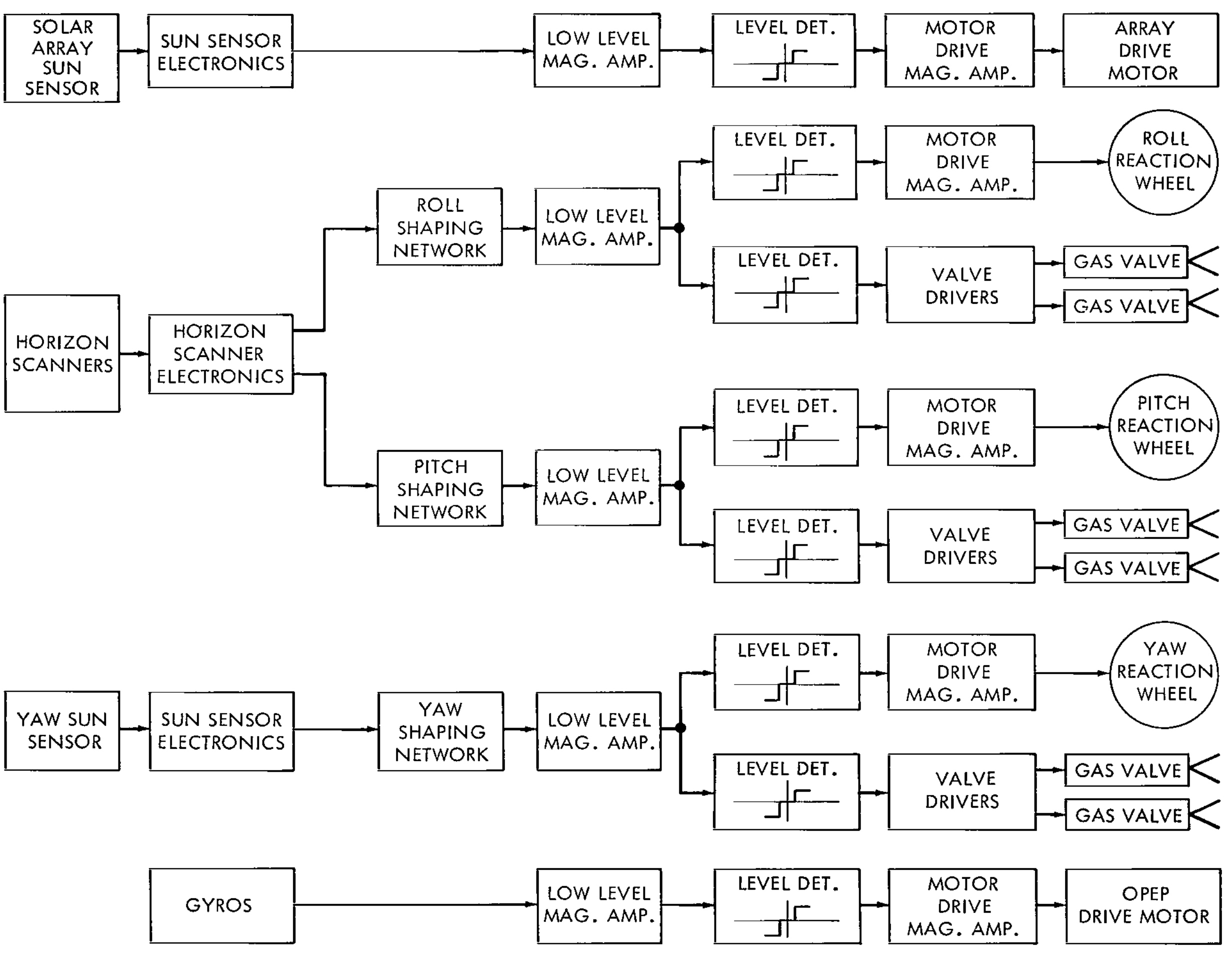

Figure 6—Simplified functional diagram of the OGO attitude control subsystem. The system is shown in its normal control mode; its configuration is somewhat different for the launch and acquisition modes, as described in the text.

respectively. The error signals from the yaw sun sensor control a reaction wheel, cold gas jet system nearly identical to the pitch system. And a drive motor rotates the array to keep that sensor error signal within its dead-band. An additional output from the fine sun sensors indicates whether or not they are illuminated by the sun. A "sun absent" indication switches control to the coarse sensors, and yaw and array torques are produced again as soon as the coarse sensors are illuminated. Thus, the system is inactive as long as the observatory is in eclipse, but reacquires the sun rapidly upon emerging from the earth's shadow.

The positions of the orbital plane experiment packages (OPEP) are controlled by a functionally independent control loop. It utilizes a single-degree-of-freedom position gyroscope operating in a gyro-compassing mode. Its angular momentum vector is perpendicular to the local vertical (earth-observatory line) and therefore to the axis of rotation of the OPEP's. The angular momentum vector is fixed with respect to the OPEP's and rotates with it. As the observatory makes each orbit around the earth, it makes one complete rotation about a line perpendicular to the orbital plane. If the gyro angular momentum vector is also aligned normal to the orbital plane, then no rotation of this vector occurs, the OPEP's are properly positioned, and no error signal developes. Whenever the gyro angular momentum vector is not normal to the orbital plane, the rotation of a component of the angular momentum vector with the observatory produces an error signal. This signal acts through a servo system, similar to the solar array system, to reposition the gyro and OPEP's so that the angular momentum vector is again normal to the orbital plane.

The control system has three modes of operation, launch, acquisition, and normal control. The launch mode, in which the control system is made inactive, is maintained until the appendages have been deployed following separation from the Agena vehicle and the observatory is illuminated by the sun. At that time the system is switched into the first phase of the acquisition mode by ground command or by the observatory sequencing equipment. In the first phase of the acquisition mode the solar array is rotated so that the solar cells face the direction away from the OPEP end of the main body. When this position has been reached, the array is held fixed with respect to the main body and the system enters the sun acquisition phase. The observatory is rotated about its yaw and roll axes to acquire the sun by the use of the error signals generated by the solar sensors. In addition, an angular rate about the pitch (longitudinal) axis of about one-half degree per second is initiated under the control of a pitch rate gyro. Sun acquisition normally requires 10 minutes or less. The earth search phase of acquisition is initiated by a timer which limits the duration of the sun acquisiton phase. During earth search the solar array continues to point toward the sun, and the main body rotates, first about the pitch axis and then about the roll axis, until the horizon scanner is locked onto the earth. Because of the small pitch rate introduced in the second phase, and because of the geometry of the orbits, each acquisition is obtained within 1 orbital period. When earth acquisition is indicated by the horizon scanners, the system switches into the normal mode of operation. The system is switched into the acquisition mode again by ground command or automatically when two or more of the four horizon scanners are not tracking.

Data Handling and Telemetry

The spacecraft data handling and telemetry subsystem is designed to process, store, and telemeter experiment and spacecraft data, and to generate timing signals for use by the experiments and the spacecraft subsystems. The major elements of the subsystem are shown in Figure 7. It is a high-capacity digital and analog system designed to condition, multiplex, store, and transmit data from the experiments and spacecraft subsystems to the ground receiving stations. Its design was based upon the requirement that the simplest practicable interface exist between the experiments and the data system. An additional consideration was the fact that the data system design had to accommodate a wide variety of experiments with, in many cases, requirements which were completely unknown at the time of design. Three forms of data from experiments can be accommodated, frequency-division-multiplexed data to the special purpose telemetry system, time-division-multiplexed analog data to the analog-to-digital converter and digital telemetry system, and time-division-multiplexed digital data directly to the digital telemetry system.

The interface wiring from a special purpose or analog experiment output to the data system consists of a single line. The requirements are specified simply, in that the output of the experiment must remain within the 0-5 volt range and have a sufficiently low output impedance that the measuring accuracy will not be unduly affected by the input impedance of the data system.

The digital data interface allows many different types of digital experiments to be flown without modification to the data system. All signal conditioning is performed within the experiments. Two types of synchronizing lines carry signals from the data system to the experiments to control their presentation of data over the digital data output lines. One type of synchronizing line provides bit pulses; the other provides word pulses for each data multiplexer input. Thus, the experimenter may divide his particular word or group of words as he desires.

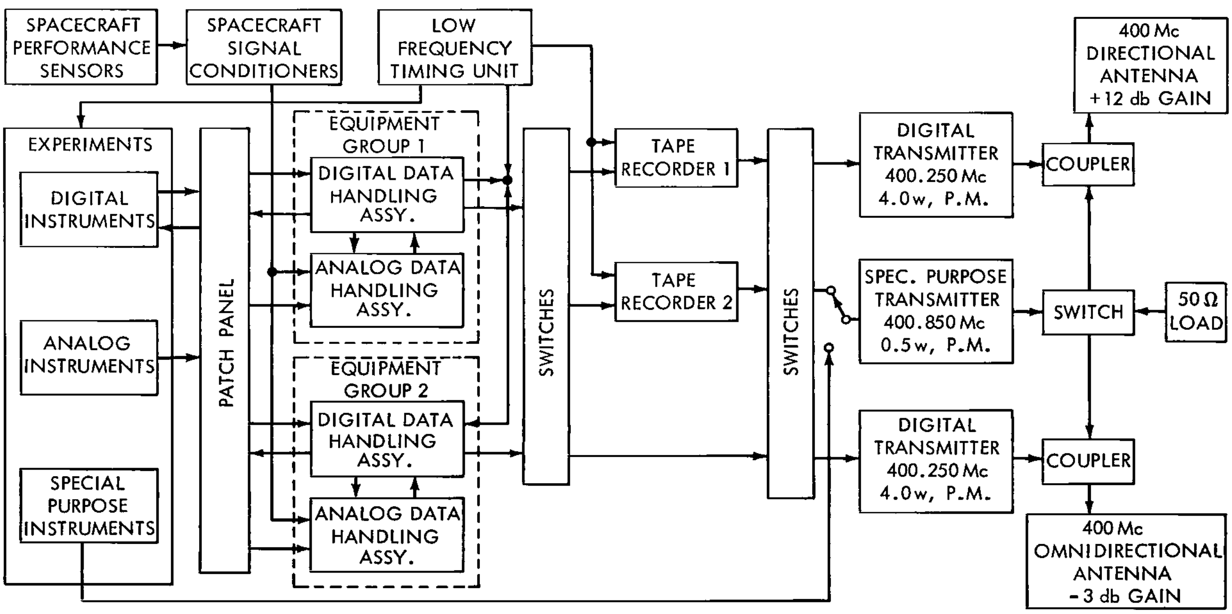

Figure 7—The OGO data handling and transmission subsystem.

Special Purpose Telemetry

The frequency-division-multiplexed special purpose system is a wideband telemetry system for experiments which are incompatible with the time sharing, or sampling, feature of the digital system. The special purpose system can accept five input signals within the frequency range of 300 cps to 100 kc, with amplitudes not exceeding 5.0 volts peak-to-peak. These signals are added in a combiner, and the composite signal amplitude is controlled by an automatic gain control circuit. The composite signal phase modulates a 400.850 Mc ±0.003 percent transmitter having a power output of 0.5 watt. The transmitter normally radiates continuously, but can be turned off and on by ground command. The special purpose transmitter feeds either the directional or the omnidirectional antenna through a command-operated coaxial switch and two 400 Mc coupler networks located in the antenna feed lines.

The signals from the experiments which are to be telemetered by the special purpose system may be of any form, as long as all the frequency components fall within the 300 cps to 100 kc band. Frequency, phase, or amplitude modulation of the signals is permissible. It is necessary that the characteristics of the five signals be chosen so that they can be separated without interference after reception on the ground. For this reason, it is usually recommended that standard IRIG frequency-modulated subcarrier oscillators be used in the experiment instrumentation whenever possible. It should be noted that the special purpose data are not stored in the spacecraft. Thus, these data are recovered only when real time telemetry is being received by the ground stations.

Digital Data Processing

Data from most of the experiments are sampled, digitized, stored, and telemetered by the wideband digital data system. As shown in Figure 7, it consists of timing assemblies to provide timing for the experiments and all of the electronic subsystems, a patch panel to facilitate connection of the experiments to the data system, data handling assemblies for sequentially sampling all data inputs and converting analog data to binary form, tape recorders for storing the binary data, and transmitters and antennas for data transmission.

The data handling system is designed to permit the greatest possible flexibility in the design of experiments. Experiments whose sensors produce basically analog signals, such as current, voltage, or resistance changes, employ signal conditioning equipment to present analog voltages in the range of 0-5 volts, with low source impedances, to the data system. Here they are converted to digital form. On the other hand, experiment sensors such as Geiger-Müller counters, etc., which produce outputs that are fundamentally digital in nature, employ digital techniques to process and condition the data. The data are presented to the data system in serial binary form in synchronism with pulses obtained from the data system.

All experiment data outputs are routed to the data handling system through a patch panel. This patch panel contains terminals for all of the experiment outputs, data system inputs, and data timing signals. The telemetry format is assembled by interconnecting these terminals. The use of the patch panel provides easy initial formating and allows last minute changes in the format without affecting the other equipment in the spacecraft or the electrical cables.

Two redundant data handling equipment groups are employed to sample the many input lines sequentially, and to convert the voltage analog signals into binary form. Normally, one group provides an output to one of the two digital transmitters for real time transmission, and the other provides an output to one of the two redundant tape recorders for storage. The roles of the two equipment groups can be reversed in the event of a partial system failure. A conceptual block diagram of an equipment group is shown in Figure 8. Although in practice the analog and digital inputs are gated in separate subassemblies, the operation is the same as though there were five time-division multiplexers. Each multiplexer is functionally equivalent to a multiple position rotary switch. The main multiplexer sequentially samples 128 inputs. Three of these inputs are outputs from the three submultiplexers, each of which samples 128 inputs. Each submultiplexer advances one position whenever the main multiplexer advances 128 positions, or one complete rotation. Thus the main multiplexer is used for rapidly varying data, and the submultiplexers are useful for sampling more slowly varying data.

Spacecraft submultiplexer 1 can be operated at the main multiplexer rate when the data from its inputs are needed more frequently, as for example during appendage deployment and initial attitude acquisition. In this event subcommutator 1 provides data directly to the transmitter or tape recorder, and the inputs to the other multiplexers are not processed. A flexible format multiplexer can be substituted for the other multiplexers on command. This device permits the time division multiplexing of 32 different data lines in 32 different sampling formats, as selected by ground command. It is provided for use when a few experiments require high sampling rates for relatively short periods of time.

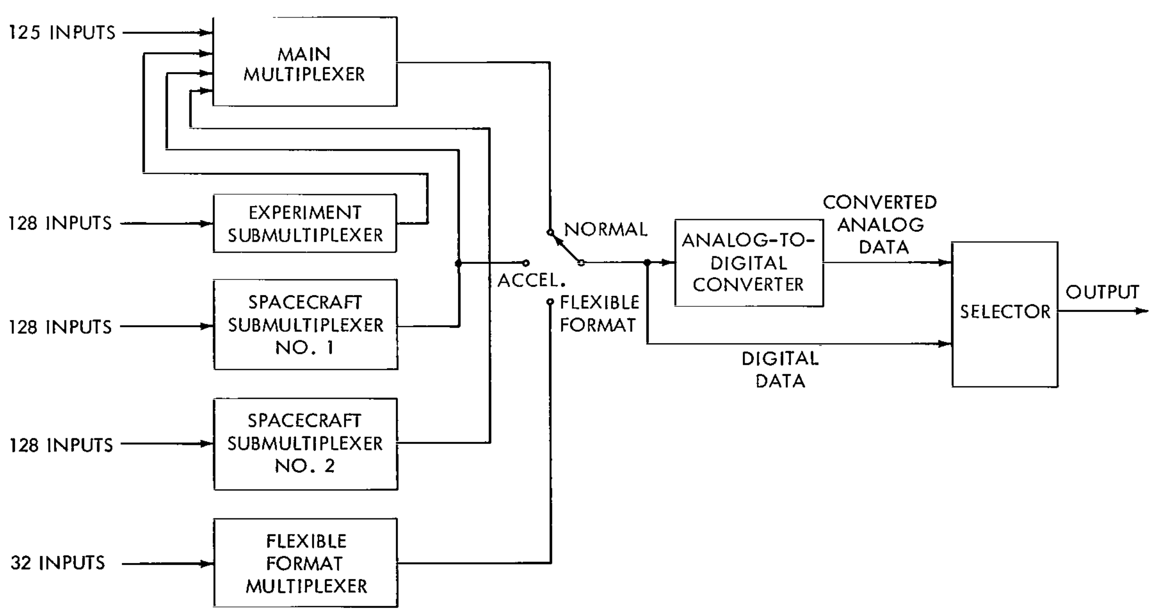

Figure 8—Block diagram of one of the two identical data handling equipment groups. The selector selects the converted analog data line whenever analog inputs are sampled, and the digital data line whenever digital inputs are sampled. (This figure is intended to give a functional picture of the system. The system is actually synthesized in a somewhat different manner.)

Each multiplexer contains both analog and digital gates, appropriately interspersed. Whenever an analog gate is turned on, the analog voltage is converted by the 8 bit analog-to-digital converter. But when a digital gate is turned on, the serial binary data bypass the converter.

The pulse code modulated (PCM) data from the data handling equipment groups are in the form of a nonreturn-to-zero (NRZ) Manchester code in which binary zeros are represented by "01" and ones by "10". This code provides at least one level transition for every bit regardless of the bit pattern, to aid in bit synchronization during ground data processing.

The largest element in the digital data format is a *sequence,* consisting of one cycling of the three submultiplexers and, thus, 128 cyclings of the main multiplexer. Each cycle of the main multiplexer, a *frame,* results in the processing of 128 words, or input *samples.* Each word consists of 9 binary *bits.* Thus, one sequence includes 1 submultiplexer cycle, 128 main multiplexer cycles or frames, 16,384 words, and 147,456 binary bits. The data handling bit rates can be set by ground command at 1000, 8000, or 64,000 bits per second for the EGO missions, and at 4000, 16,000, or 64,000 bits per second for the POGO missions. Tape recording in the observatory is always done at 1000 or 4000 bits per second, and tape recorder readout occurs at 64,000 or 128,000 bits per second, for EGO and POGO respectively. Real time digital telemetry can occur at any of the bit rates, depending on the requirements of the experiments. These format specifications and bit rates result in the sampling of each of the 128 inputs to the main multiplexer 0.8681, 3.4724, 6.9448, 13.8896, or 55.5584 times per second and of each of the 384 inputs to the three submultiplexers every 147.456, 36.864, 18.432, 9.216, or 2.304 seconds, depending on whether the 1, 4, 8, 16, or 64 kilobit rate is in use.

Additional signals available to the experiments include power converter 2461 cps synchronization, motor 400 cps synchronization, ground commands, and timing at 0.01, 0.1, 1, 10, 100, 1000 pulses per second. Timing pulses corresponding to the sampling times of many of the digital inputs are provided to assist the experimenters in programming the data conditioning within their experiments. To assist the experimenter in determining the data handling system operating conditions, additional signals indicate whether real time data are being transmitted, the real time bit rate, and the equipment group which is feeding the data storage system.

Digital Data Storage

Two identical, redundant tape recorders store the digital data so that continuous data can be received from the observatory by a small number of ground stations. Each of the recorders has a storage capacity of 43.2 million binary bits. The recording bit rate is either 1000 or 4000 binary bits per second, depending on the mission; thus, the recorders can record for 12 or 3 hours, respectively. The two recorders can store sequentially to provide up to either 24 or 6 hours between readouts. Readout of one recorder can occur while data are being stored on the other, to provide continuous coverage. Readout times for the two cases are 11.25 and 5.625 minutes, respectively. The recorder tapes are reversed for readout, resulting in time reversal of the data. Time reverses again during processing on the ground, returning the data to their original order.

Digital Data Telemetry

The digital outputs of either of the two data handling equipment groups or either of the two tape recorders are telemetered on ground command by either of the transmitters. Complete command-controlled cross-strapping provisions allow the full use of the extensive parallel redundancy to increase the reliability of the data handling system.

One of the two digital wideband transmitters is energized upon the receipt of a ground command. The telemetry system is automatically turned off by a timer approximately 23 minutes after the loss of the command carrier. One of the transmitters feeds the omnidirectional antenna, which has a gain of -3 db in the earthward hemisphere, relative to isotropic radiation, and is circularly polarized. The other digital transmitter drives the directional antenna which has a gain of +12 db, has a half-power beamwidth of less than 40 degrees, and is circularly polarized. Normally the transmitter driving the directional antenna will be used when the transmission distance is greater than about 3 earth radii. When the observatory is near the earth the omnidirectional antenna with its greater beamwidth will be used. It is not possible for both digital transmitters to operate simultaneously, but one digital transmitter and the special purpose transmitter may transmit concurrently. If both digital transmitters should fail, or if a lower transmitter power is desired, then the digital data can be transmitted by the special purpose transmitter.

The digital wideband transmitters have power outputs of 4 watts. The 400.250 Mc ±0.003 percent carriers are biphase modulated by the PCM data. The angle between the two phases is adjusted to leave approximately 10 percent of the radiated power at the carrier frequency. This simplifies lock-on and tracking of the carrier by the ground receivers.

Observatory Synchronization and Timing

A central timing system provides highly accurate timing and synchronization for the entire observatory. The basic timing sources are two redundant 256 kc crystal oscillators having long term stabilities of one part in 10^5 per year and short term stabilities of one part in 10^6 per hour. Only one oscillator is used at a time so that all timing is derived from a single source. Countdown circuits produce signals for synchronizing the data handling assemblies and the tape recorders, for time reference in the experiments, and for synchronizing all power converters to minimize interference to VLF experiments. An additional register generates observatory accumulated time, which is recorded and telemetered with all digital data to serve as a basic data-time reference.

Ground Command Reception

Two redundant AM command receivers operating at approximately 120 Mc are fed from dipole omnidirectional antennas (Figure 9). The dipoles are crossed in a single assembly, thus

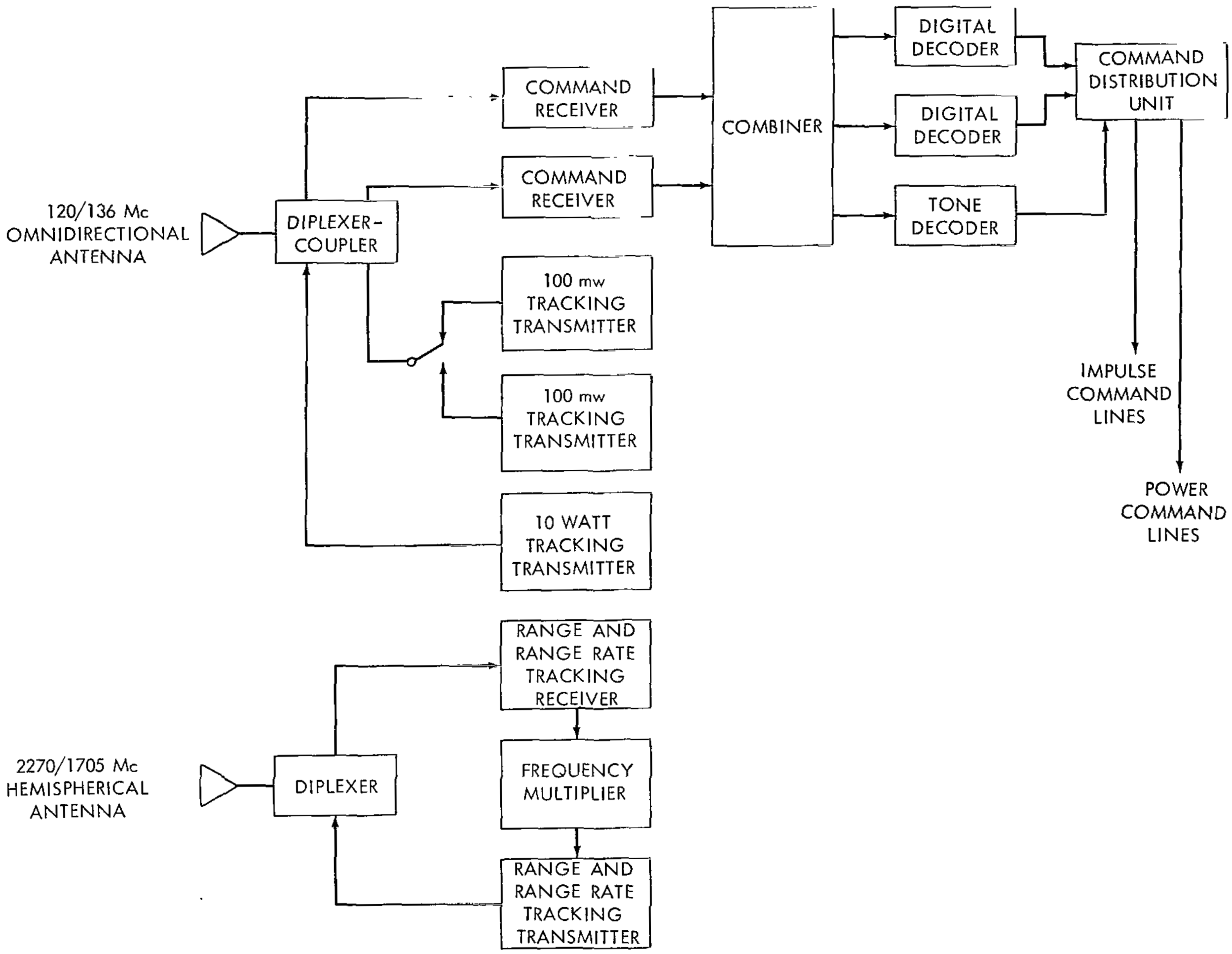

Figure 9—Functional diagram of the OGO command and tracking subsystem.

providing polarization-diversity reception. The receivers have 33.15 and 7.3 Mc IF frequencies, and IF bandwidths of 40 kc. The bandwidths of the audio sections are 11 kc. The basic receiver noise figures are 4 db. With an antenna noise temperature of 1000°K, the command noise power is -121 dbm. The receivers are set to unsquelch at -115 dbm, and at the same point relays operate to indicate the presence of an RF carrier. Each receiver contains two AGC loops to permit operation over a wide range in signal strengths.

The outputs of the two command receivers feed, in a parallel redundant fashion, two digital decoders and a single tone decoder. The squelch or failure detection circuits in the receivers maintain the input to the decoders at a constant level, regardless of the number of receivers operating.

The digital decoders permit the reception and proper routing of 254 independent commands. They operate on a frequency shift keying (FSK) signal where one frequency represents a binary "0" and a second represents a "1". Each digital decoder can be addressed separately, and the output from a single decoder provides complete digital command capability. Outputs from the digital decoder operate relays arrayed in a 16 by 16 matrix. Two types of relays are used, power command and impulse command. Of the 254 commands, 104 are utilized to control the data handling, communications, power, attitude control, and thermal systems and to initiate deployment of the appendages. The other 150 commands are reserved for the experiments. Fifty power relays, requiring separate on and off commands, provide electrical power to the various experiments. And 50 impulse relays provide the grounding of 50 control lines for approximately 50 msec, following the execution of the proper commands.

The digital command words contain 24 binary bits. The first bit is always "1" to provide synchronization. The next 3 bits contain the satellite and decoder addresses. The following 2 bits designate the mode of operation of the decoder, and the next 8 bits contain the command itself and select the proper relay in the command distribution unit. The complement to the 2 mode bits and 8 command bits is retransmitted as a parity check. If the parity check succeeds, a command-execute signal starts the proper command relay, and command execution is indicated in the telemetered data.

A few of the most important commands can be received as tone commands and decoded in the relatively simple and highly reliable tone decoder. This sequential tone command system permits the reception of real time digital data from the observatory at secondary receiving stations without requiring that they have the somewhat complex digital command generator. In addition, this simple tone command system permits limited operation of the observatory and recovery of data in the event of failure of the digital command system.

Observatory Tracking Equipment

The orbits of most previous satellites have been determined by the use of the World-Wide Tracking Network, formerly known as the Minitrack Network. This network of tracking stations and the necessary computational techniques are well established, and will be used in the OGO program. The system will be supplemented by a range and range rate system which is expected to permit more accurate computation of the orbit parameters in a shorter time, especially in the case of a highly eccentric orbit in which the satellite spends a large fraction of its time at large distances from the earth and the angular rates are very low.

The observatory tracking system components are indicated in Figure 9. Three 136.00 ± 0.41 Mc beacon transmitters will provide a continuous tracking signal for the ground stations. One of the two redundant low power (100 mw) transmitters operates continuously, except when the high power (10 watt) transmitter is energized. The high power transmitter, utilized only on missions with apogee distances greater than approximately 2 earth radii, is controlled by a timer which turns off the transmitter 45 seconds after it is energized.

The beacon transmitters use the same crossed dipole omnidirectional antenna as the command receivers. A diplexer-coupler provides the necessary isolation between the beacon transmitter and the receivers. For beacon transmission the antenna polarization is circular.

The completely independent range and range rate system utilizes a diplexed antenna, receiver, frequency multiplier, and transmitter. Signals at frequencies of approximately 2270 and 2271 Mc are received from two ground stations simultaneously, converted, and retransmitted as 1.4 and 3.2 Mc sidebands on a 1705.000 Mc carrier. The received signals are phase modulated by range tones at frequencies of: (1) 500, 100, or 20 kc, (2) 4 kc, (3) 800 cps, and (4) 160, 32, or 8 cps. The ground stations determine the range of the observatory by comparing the phases of the transmitted and received modulating frequencies. The range rate is ascertained by measuring the doppler shifts of the RF signals. The use of two ground tracking stations simultaneously permits highly accurate trilateration of the observatory.

The overall goal of the tracking program is to be able to determine for the experimenters the position of the observatory, at all times, within a sphere of uncertainty having a radius of 1 km at radial distances of less than 1000 km and a radius of 100 km at radial distances of 17 earth radii.

THE OGO EXPERIMENTS

Experiments for the Orbiting Geophysical Observatories are selected by the Office of Space Sciences, NASA Headquarters, Washington, D. C. from those proposed by research groups in the universities, industry, NASA laboratories, and other government agencies. Many of the experiments are the result of initial technical development supported by NASA as a part of its advanced development program. Selection of experiments has been completed for the first EGO and POGO launches. Lists of these experiments, the principal investigators, and their institutions are included in Tables 1, 2, and 3. More complete technical descriptions of these experiments will be published from time to time by the experimenters.

OGO ORBITS

The Orbiting Geophysical Observatories can be placed in a number of orbits, depending on the needs of the experiments. The minimum perigee height is set by the requirement that the atmospheric drag must be small enough to ensure a 1 year lifetime. The maximum apogee height is limited at the present time to about 24 earth radii (geocentric), for a 31 degree inclination, by the capabilities of presently available launch vehicles. As larger launch vehicles become available, missions with apogee at lunar distances will require only minor adjustments of some of the subsystem parameters. The orbital inclination is limited by the launch site and range instrumentation locations. The Eastern Test Range at Cape Kennedy, Florida, is located at a geographic latitude of about 28.5 degrees. Thus, orbital inclinations between 28.5 and approximately 50 degrees are achievable without flying over inhabited land masses during the initial trajectories. Lower orbital inclinations are possible only by altering the course of the vehicle after lift-off or by launching from a lower latitude. Higher inclination orbits, including polar orbits, are possible by launching the observatory from the Western Test Range at Lompoc, California.

Table 1

Experiments for OGO-A and OGO-B (EGO Orbits).

Experimenter	Experiment	Detector
K. A. Anderson University of California	Solar proton flux, 10-90 Mev, energy and variations	Scintillation counter
J. H. Wolfe Ames Research Center	Solar plasma flux, energy and direction	Electrostatic analyzer
H. J. Bridge Massachusetts Institute of Technology	Solar plasma flux, energy and direction	Faraday cup
T. L. Cline and E. W. Hones Goddard Space Flight Center and Institute for Defense Analysis	Search for positrons and solar gamma ray flux and spectrum	Scintillation counters
A Konradi Goddard Space Flight Center	Geomagnetically trapped electron and proton flux, energy and direction	Phosphor scintillation counter
F. B. McDonald and G. H. Ludwig Goddard Space Flight Center	Galactic and solar cosmic ray flux, charge and energy	dE/dx vs. E scintillation telescope
J. A. Simpson University of Chicago	Galactic and solar cosmic ray flux, charge and energy	dE/dx vs. E and range detector
J. A. Van Allen State University of Iowa	Geomagnetically trapped electron and proton flux and energy	Omnidirectional Geiger counters and solid state detector
J. R. Winckler and R. L. Arnoldy University of Minnesota	Geomagnetically trapped electron energy and flux, and total ionization	Magnetic electron spectrometer and ion chamber
R. S. Lawrence National Bureau of Standards	Electron density by RF propagation, 40 and 360 Mc	Radio transmitter
R. Sagalyn Air Force Cambridge Research Laboratory	Thermal charged particle density, energy, and composition	Spherical ion and electron trap
E. C. Whipple Goddard Space Flight Center	Thermal charged particle density, energy, and composition	Planar ion and electron trap
H. A. Taylor Goddard Space Flight Center	Atmospheric composition, 1-48 amu	Bennett RF mass spectrometer
J. P. Heppner Goddard Space Flight Center	Magnetic field strength and direction	Rubidium-vapor and flux-gate magnetometer
E. J. Smith Jet Propulsion Laboratory	Magnetic field low frequency variations, 0.01-1000 cps	Triaxial search coil magnetometers
W. M. Alexander Goddard Space Flight Center	Micron dust particle velocity and mass	Time-of-flight and momentum detector
F. T. Haddock University of Michigan	Solar and Jovian radio-noise burst frequency spectrum, 2-4 Mc	Radio receiver
R. A. Helliwell Stanford University	VLF terrestrial noise, solar particle emissions, and cosmic noise frequency distribution and strength, 0.2-100 kc	VLF receiver
P. Mange Naval Research Laboratory	Geocoronal lyman-alpha intensity and location of scattering layer	Lyman-alpha ion chambers
C. L. Wolff, K. L. Hallam and S. P. Wyatt Goddard Space Flight Center and University of Illinois	Gegenschein intensity and location	Gegenschein scanning photometer

Table 2

Experiments for OGO-C and OGO-D (POGO Orbits).

Experimenter	Experiment	Detector
R. A. Hoffman, L. R. Davis, A. Konradi, and J. M. Williamson Goddard Space Flight Center	Low-energy trapped radiation; electrons, 10-100 kev; protons, 100 kev-4.5 Mev	Phosphor scintillation counter
H. V. Neher and H. Anderson California Institute of Technology and Jet Propulsion Laboratory	Total ionization over the polar regions	Ionization chamber
J. A. Simpson University of Chicago	0.3-30 Mev nucleons	Scintillation telescope
J. A. Van Allen University of Iowa	Net downflux of corpuscular radiation in auroral zones and over the polar caps	Geiger counters
W. R. Webber University of Minnesota	Energy spectrum and charged-particle composition of galactic and solar cosmic rays	Scintillation Čerenkov detector
R. E. Bourdeau Goddard Space Flight Center	Ionospheric charged particles	Planar retarding potential analyzer
L. M. Jones and E. J. Schaefer University of Michigan	Neutral-particle and ion measurements: 0-6 and 0-40 amu	Paul massenfilter mass spectrometer
G. P. Newton Goddard Space Flight Center	Neutral-particle density	Bayard-Alpert density gauge
H. A. Taylor and H. C. Brinton Goddard Space Flight Center	Atmospheric composition 1-45 amu	Bennett RF mass spectrometer
W. M. Alexander, C. W. McCracken, O. E. Berg, and L. Secretan Goddard Space Flight Center	Micrometeorites: mass, velocity, charge	Time-of-flight and momentum detector
J. P. Heppner, H. R. Boroson, and J. C. Cain Goddard Space Flight Center	World Magnetic Survey	Rubidium-vapor magnetometer
R. E. Holzer and E. J. Smith University of California at Los Angeles and Jet Propulsion Laboratory	Magnetic field fluctuations, 1-1000 cps	Triaxial search coil magnetometer
F. T. Haddock University of Michigan	Radio astronomy measurements of galactic emission at 2.5 and 3.0 Mc	Radio receiver
R. A. Helliwell Stanford University	VLF measurements at 0.2-100 kc	VLF receiver
M. G. Morgan and T. Laaspere Dartmouth College	VLF emissions and whistlers between 0.5 and 10 kc	VLF receiver
C. A. Barth and L. Wallace Jet Propulsion Laboratory, Yerkes Observatory	Measurements of airglow, 1100 to 3400A	Ebert UV spectrometer
J. Blamont and E. I. Reed University of Paris and Goddard Space Flight Center	Airglow in the UV and at 3914, 5577 and 6300A	Photometers
H. E. Hinteregger Air Force Cambridge Research Laboratory	Solar emission in the 200-1600A region	Scanning spectrometer
R. W. Kreplin, T. A. Chubb and H. Friedman Naval Research Laboratory	Solar x-ray emissions in the 0.5-3, 2-8, 8-16, and 44-60A bands	Ionization chambers
P. M. Mange, T. A. Chubb and H. Friedman Naval Research Laboratory	Lyman-alpha and far UV airglow between 1230 and 1350A	Ionization chambers

Table 3

Experiments for OGO-E (EGO Orbit).

Experimenter	Experiment
R. L. F. Boyd and A. P. Willmore University College London	Electron temperature and density
R. Sagalyn and M. Smiddy Air Force Cambridge Research Laboratory	Thermal and epithermal plasma
G. P. Serbu and E. J. Maier Goddard Space Flight Center	Electron and ion measurements (0 to 100 ev)
K. A. Anderson and H. Mark University of California	Energetic solar flare radiation X ray and ionizing radiation
T. L. Cline Goddard Space Flight Center	Low-rigidity interplanetary electrons, positrons, and protons, and gamma rays
R. D'Arcy, L. G. Mann and H. I. West, Jr. Lawrence Radiation Laboratory	Electron and proton spectrometer
L. A. Frank, W. A. Whelpley and J. A. Van Allen University of Iowa	Low energy electron detector
G. W. Hutchinson, D. Ramsden, and R. D. Wills University of Southampton	Energetic phons in primary cosmic ray photons
P. Meyer and C. Y. Fan University of Chicago	Cosmic ray electrons (20 to 100 Mev)
F. B. McDonald, G. H. Ludwig, D. E. Hagge and V. K. Balasubrahmanyan Goddard Space Flight Center	Galactic and solar cosmic rays
K. W. Ogilvie and T. D. Wilkerson Goddard Space Flight Center and University of Maryland	Tri-axial electron analyzer (0-15 kev)
A. H. Wapstra, Y. Tanaka, M. N. Lund, and I. Scheepmaker Delft Technical University	Cosmic ray electrons (0.5 to 10 bev)
P. J. Coleman, T. A. Farley, and D. L. Judge University of California at Los Angeles and University of Southern California	Tri-axial fluxgate magnetometer and trapped particles
J. P. Heppner, T. L. Skillman, B. G. Ledley, M. Campbell, and M. Sugiura Goddard Space Flight Center	Magnetic field measurements
E. J. Smith and R. E. Holzer Jet Propulsion Laboratory and University of California at Los Angeles	Search coil magnetometer
C. W. Snyder, M. Neugebauer and J. L. Lawrence Jet Propulsion Laboratory	Plasma spectrometer
G. W. Sharp and T. J. Crowther Lockheed Missiles and Space Company	Light ion magnetic mass spectrometer
W. M. Alexander, O. E. Berg, C. W. McCracken and L. Secretan Goddard Space Flight Center	Micrometeorites
F. T. Haddock University of Michigan	Radio astronomy (0.05 to 2 Mc)
C. A. Barth and J. B. Pearce Jet Propulsion Laboratory	UV photometric measurements Solar UV photometer
J. E. Blamont University of Paris	Geocoronal hydrogen
To be selected	Solar UV photometer

Table 4

Nominal Initial Orbital Parameters.

Parameter	EGO	POGO
Semi major axis	62,450 km	6970 km
Eccentricity	0.8934	0.04830
Inclination	30.8 deg	90 deg
Argument of perigee	-45.8 deg	-73.8 deg
Right ascension of ascending node	144.5 deg	-19.4 deg
Period	43.143 hr	1.609 hr
Injection geodetic latitude	-20.4 deg	-17.1 deg
Injection longitude	111.9 deg	47.1 deg

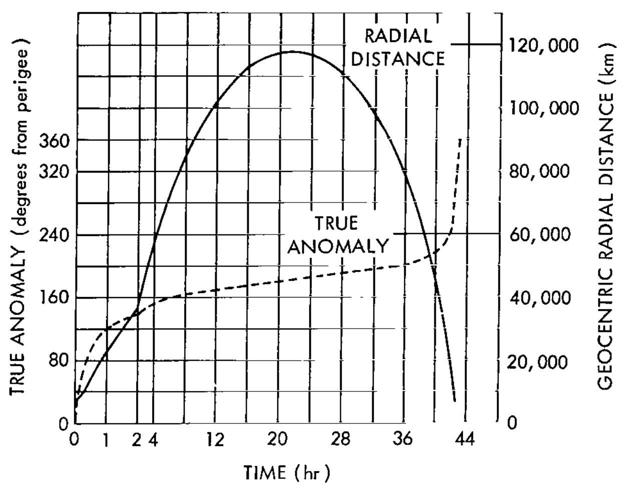

Figure 10—True anomaly and geocentric radial distance for the EGO orbit. The semimajor axis is 62,451 km and the eccentricity is 0.8934. The observatory will spend about 75 percent of its time beyond the magnetosphere.

The planned nominal orbital parameters for the EGO orbit are listed in Table 4. This orbit was chosen to enable the observatory to traverse the high intensity radiation belts for a study of their characteristics throughout the entire trapping region, and to make atmospheric, exospheric, ionospheric, and magnetospheric measurements from near the earth to interplanetary space. Especially interesting is the possibility of studying the shape of the boundary of the magnetosphere and of the plasma shock front presumed to exist on the sunward side of the boundary. A number of interplanetary experiments are included, since the observatory will spend long periods of time beyond the magnetosphere.

The characteristics of this orbit can be seen with the use of several graphs. For the purposes of most of the orbit calculations the middle of the fourth quarter, 2000 UT on November 15, 1963, was used as a launch time in the evaluations. Adjustments will be necessary when the exact date and time are chosen. Figure 10 indicates the geocentric radial distance and the true anomaly as a function of time for the initial orbit. The true anomaly is the perigee, earth center, observatory angle. The percentage of time that the observatory will spend within a specified range in height can easily be ascertained from this graph.

The region in geomagnetic space through which the observatory will pass during its first month in orbit is indicated in Figure 11. The orbit is projected on a meridian plane whose coordinates are geocentric radial distance and geomagnetic latitude. The first orbit is plotted to indicate the general form of the orbits. The bands span 22.8 degrees in latitude (twice the angle between the geomagnetic and geographic equatorial planes), and the satellite ranges between plus and minus (i + 11.4) degrees in latitude where i is the orbital inclination. For this particular EGO orbit, i increases from its initial value of 30.8 degrees to 42.2 degrees by the end of 1 year because of the orbital perturbations induced by the gravitational fields of the sun and moon. This coupling also causes the perigee height to increase from 277 to 3164 km by the end of the year. (These orbital perturbations depend strongly on the time of launch.)

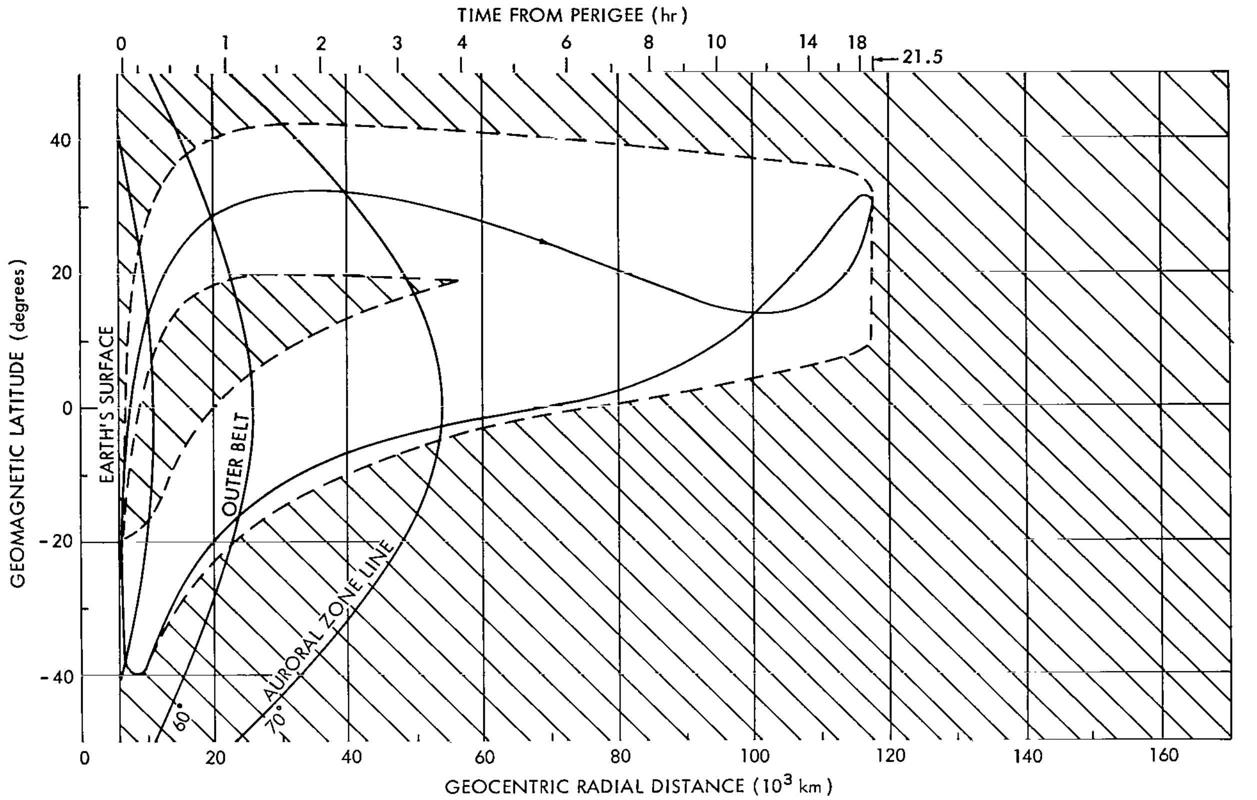

Figure 11—Nominal geocentric coordinate limits for the first month of the EGO. The semimajor axis is 62,451 km, the eccentricity is 0.8934, and the inclination is 30.766 degrees. The first complete orbit is indicated by the heavy line. The center of the inner proton radiation belt is located just inside the 40 degree magnetic field line; the outer belt is centered just inside the 60 degree field line. The centered dipole geomagnetic field model was used.

The lines-of-sight of experiments looking away from the earth will trace a simple path on the celestial sphere, as shown in Figure 12. Paths are shown for the initial orbit and for the last orbit in the first year. The shaded area contains all paths throughout the first year after the launch. The path changes phase throughout the year because of the precession of the right ascension of the ascending node from 144 degrees initially to 115 degrees after 1 year. The path extends to a higher declination because of the increase in orbital inclination mentioned before. Changes in the launch time and date will shift the phase of the paths shown.

The path of the sun, and therefore the path of the line-of-sight of experiments looking toward the sun throughout the 1 year period, is also shown in Figure 12. On March 21 the sun is located at zero degrees RA and zero degrees dec. It proceeds along the ecliptic plane line at approximately 0.98 degree per day.

The approximate orbital parameters for the first POGO are also listed in Table 4. This orbit will allow the observation of many phenomena directly over the polar and auroral regions, and the determination of the variations of these phenomena over the complete range in latitude.

For each mission the time of launch, and therefore the location of the orbit in space, must be chosen to satisfy a number of limiting conditions. These launch window restraints include the following:

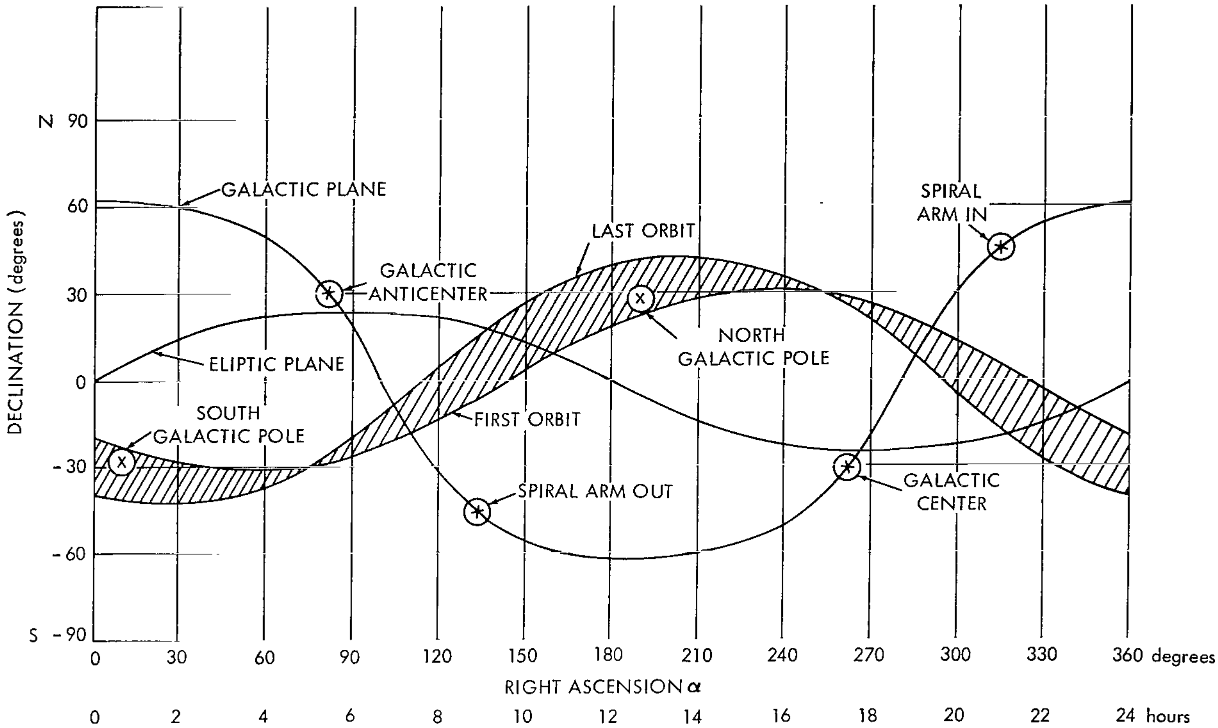

Figure 12—Projection of the lines-of-sight looking away from the earth and toward the sun on the celestial sphere. The initial inclination is 30.766 degrees, the initial argument of perigee is −45.774 degrees, and the right ascension of the ascending node is 144.456 degrees. The detectors observing the sun trace a path around the celestial sphere during a year, along the path marked "ecliptic plane."

1. For a high eccentricity orbit such as the EGO orbit, the gravitational coupling of the observatory with the sun and moon is significant, resulting in a continuous change of all orbital parameters including the perigee height. The observatory must be launched so that the perigee height will never drop below approximately 230 km, to insure that it will not lose its kinetic energy because of aerodynamic drag and plunge into the earth.

2. The maximum time per orbit that the observatory lies within the earth's shadow must not exceed 2 hours during the 1 year operating period. For longer eclipse times the temperature of the solar array would drop to below -140°C and damage to the solar cell mountings would result from stresses produced by the unequal thermal contractions of different materials.

3. During the launch, deployment, and initial acquisition sequences, the sun must not directly illuminate the louvered thermal control surfaces for extended periods of time.

4. The experiments may impose additional restraints. For example, it may be desirable to specify an initial value for the angle between the line of apsides and the earth-sun line.

These restraints require very extensive orbital calculations in selecting a suitable launch date and time. These studies may indicate the need for modification of certain restraints to permit a launching in the desired season. For example, it may be necessary to include a timer in the observatory to delay the beginning of the attitude control acquisition cycle, to permit orbital injection in the earth's shadow.

PRELAUNCH OPERATIONS

Between the times that the experiment fabrication is completed and the observatory is launched into orbit, it is necessary to electrically check and environmentally test the experiments, integrate them into the spacecraft, calibrate them, check for interference, environmentally test the complete observatory, ship the observatory to the launch site, and conduct the launch site operation. These steps are necessary to ensure that the experiments and spacecraft operate properly as an integrated unit, that they are properly calibrated, and that they will have a reasonable probability of surviving the conditions imposed by the launch and space environments. These steps are expected to require approximately 1 year, at least for the first few observatories.

The OGO testing philosophy attaches great importance to the complete testing of assemblies before they are integrated into the observatory, to detect most of the design and production errors and early component failures. Environmental tests of the completely integrated observatory are also made, primarily to uncover design and fabrication errors in the interconnection of assemblies and the mechanical attachment of the assemblies to the spacecraft structure, to disclose problems which result from interactions between assemblies, and to detect additional early component failures.

Electrical, Mechanical, and Thermal Testing of Experiments

Whenever a new experiment assembly is designed for use in the OGO program, the first flight-quality unit, usually referred to as the prototype unit, is subjected to a set of electrical and environmental design qualification tests to ensure that the mechanical, electrical, and thermal designs have a sufficiently large margin of safety. A safety factor of 1.5 usually is required for all mechanical conditions, and all assemblies are tested over a temperature range which extends 20°C greater than than anticipated in orbit. After the design has been qualified, at least two units, a flight unit and a spare flight unit, are built for each launch attempt. These units are subjected to a set of flight acceptance tests at levels equal to those expected during the actual launch and operation in orbit. The tests are intended primarily to disclose poor workmanship and to induce early component failures.

Electrical Tests

Each experimenter is required to furnish ground checkout equipment with his experiment instrumentation. Generally this equipment is built in two parts. The first part is a sensor exciter which provides an input to the basic sensor or detector. It may be very simple, such as a radioactive source for checking G.M. counters, or somewhat more complex, such as a precision variable low-current source for calibrating the input electrometer circuits of ion collectors. Its function is to produce a predictable nonzero output of the detectors so that an end-to-end check of the operation of the instruments can be made periodically throughout the testing program, and to facilitate detailed calibration of the experiments.

The second part of the ground checkout equipment is used for more extensive testing and calibration of the instruments. This experiment test set is capable of operating the experiment independently of the spacecraft, and is used by the experimenter to check, calibrate, or find difficulties in the instruments. It may include some or all of these provisions:

1. It may provide all signals to the experiment which are normally obtained from the spacecraft in flight, such as bit pulses, word pulses, switch mode and bit rate signals, timing pulses, command and synchronizing signals, and dc power, so that the experiment instrumentation may be operated independently of all other equipment.
2. It may have a means for accepting data from the experiment and displaying them.
3. It may be capable of monitoring a number of test parameters, such as internal voltages, waveforms, and event occurrences.
4. It may provide calibration inputs; for example, a means for substituting pulses of known characteristics at some internal point in the instrumentation.
5. It may have control inputs. An example is the relaxation of coincidence requirements in an energetic particles directional telescope to facilitate ground checkout by particles from a radioactive source having energies too low to penetrate both detectors.

The sensor exciter will be used throughout the observatory integration, testing, and prelaunch operations to check the operation of the experiments. The experiment test set normally will not be used after the experiments are mounted in the spacecraft except for detailed experiment calibration, interference detection, and troubleshooting in the event of a failure during the testing program.

Upon delivery to Goddard Space Flight Center, the experiments are connected to a spacecraft electrical simulator. This equipment presents an electrical interface to the experiments which is nearly identical to that presented by the spacecraft. It is capable of providing bit, word, switch, mode, bit-rate, timing, command, and power synchronization signals and electrical power to the experiments and accepting and processing data from experiments, either singly or in groups of any size up to the full complement for a particular mission. Provisions are included to vary critical parameters, such as the supply voltage, pulse amplitudes, etc. to ensure that the electrical performance of the experiments is nonmarginal.

The spacecraft simulator is shown in block diagram form in Figure 13. It contains a low frequency timing assembly, digital data handling assembly, and analog data handling assembly electrically identical to those used in the spacecraft. The analog data handling assembly is also mechanically identical to the one used in the spacecraft, since the distributed capacitances and lead lengths are somewhat critical in this assembly. The spacecraft simulator contains additional assemblies for performance monitoring purposes and to provide signals which are not derived from the data handling assemblies.

The method of using the spacecraft simulator for experiment testing is shown in Figure 14. The experimenters' sensor exciters and test sets are used to stimulate the detectors and to monitor and control the experiments as before. The spacecraft simulator provides the necessary driving signals to the experiments and accepts their data. The PCM ground station demultiplexes the data. A number of display devices, including a multiple pen analog recorder, a multichannel optical oscillograph, a bar oscilloscope, and a digital printer are included as a part of the ground station. A medium size digital computer (Scientific Data Systems model 910) is included as a part of the experiment checkout system for use when more complex sorting, rapid access storage, computation, and output of data is desired. This computer has a core storage capacity of 4000 twenty-four bit words, and a priority-interrupt feature which allows the computer to accept data from several sources and to operate on several programs concurrently. Thus, completely independent processing programs can be run on several different experiments at the same time.

A number of these electrical test systems have been or are being assembled for location at the experiment electrical, mechanical, and thermal test areas, the observatory integration and test areas, and the launch sites. The principal purposes of the systems are:

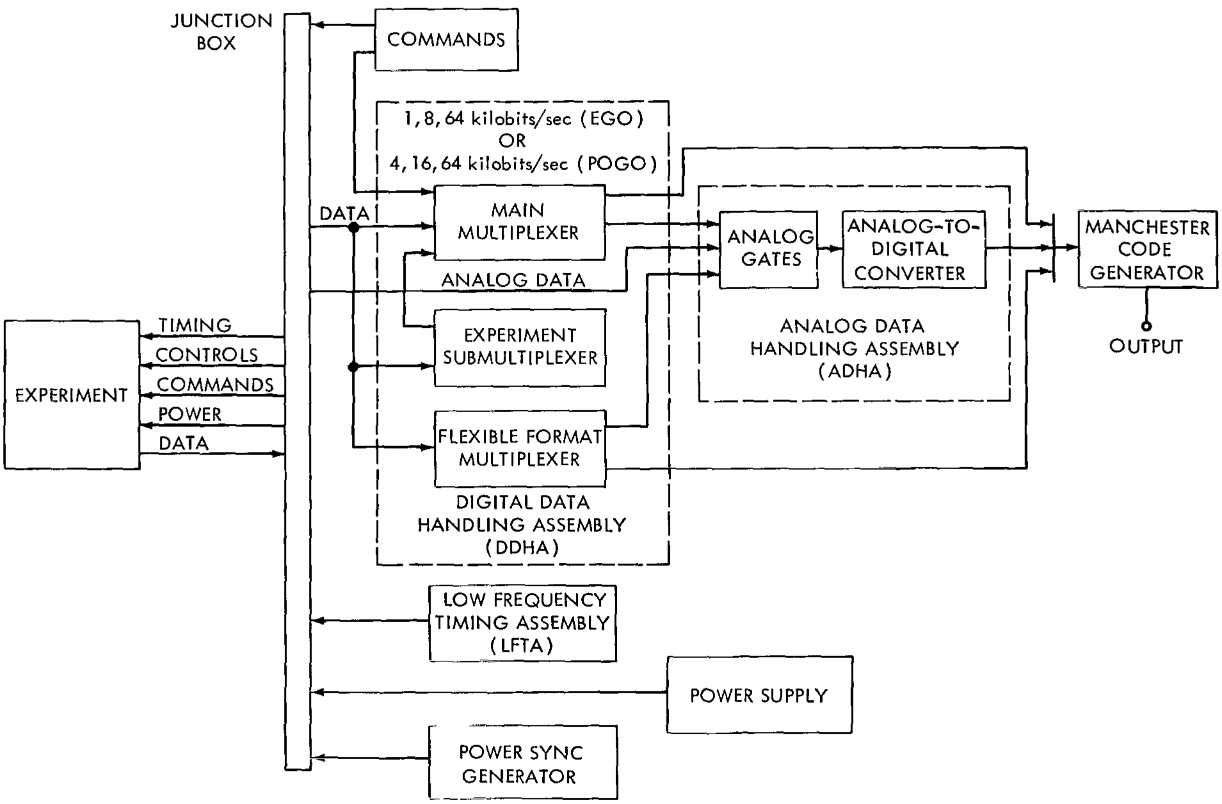

Figure 13—The spacecraft simulator for checking the operation of experiments when not mounted on spacecraft.

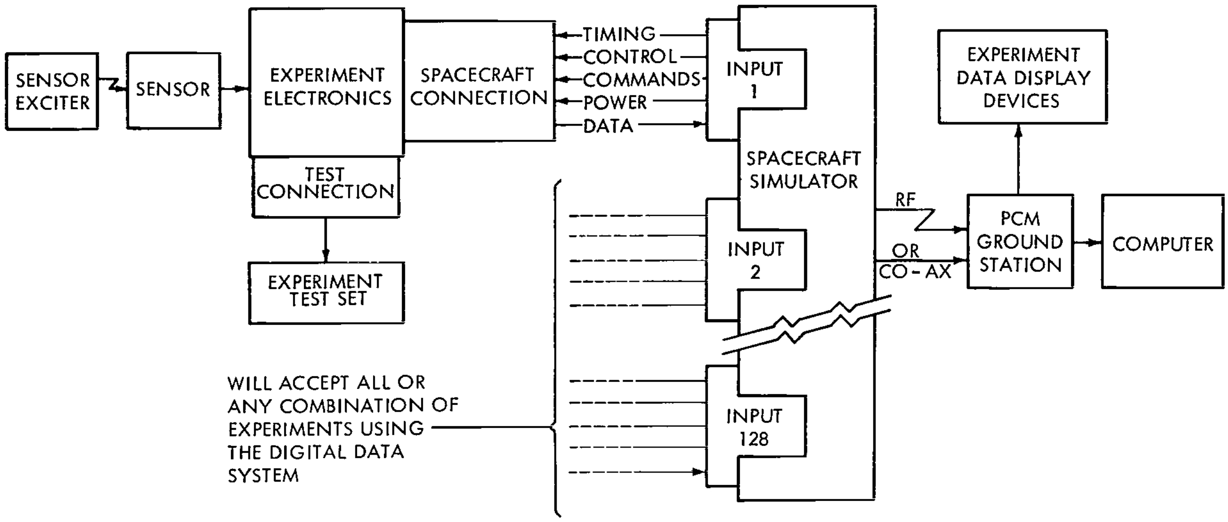

Figure 14—The use of the experimenter's test equipment, spacecraft simulator, PCM demultiplexing equipment, and computer in checking and calibrating OGO experiments.

1. For the use of the experimenters in checking the operation of the breadboards of experiments. The earlier an experiment is tested on a realistic interface, the easier it is to make necessary corrections.

2. The checking of the electrical performance of flight quality experiment assemblies as soon as they are completed, in order that design and fabrication errors may be detected.

3. For the use of the experimenters in calibrating their instruments.

4. For the use of the experimenters in locating difficulties in their instruments after failures. Whenever improper operation of any experiment assembly in the spacecraft is detected, the experiment will be removed from the spacecraft for analysis with the use of the simulator or the experimenter's checkout equipment. It will be returned to the spacecraft after it has been repaired.

5. Data from the observatory will also be processed by the PCM ground station and computer in checking experiments when they are mounted and operated in the spacecraft. Thus, the same data processing programs and procedures can be used for observing the operation of the experiments throughout the entire program.

Mechanical Tests

The prototype and flight experiment and spacecraft assemblies, and the complete observatories, are subjected to mechanical environmental tests to detect design and fabrication errors in both the components and the assemblies. In general, the test levels are 1.5 times the maximum expected launch conditions for the prototype assemblies and observatories and are equal to the expected launch conditions for the flight assemblies and observatories. The tests are as follows:

1. Measurements are made of the critical dimensions of the assemblies and the locations of mounting provisions, detector openings, etc., to assure that the equipment will fit into the spacecraft. The weights and centers of gravity are also measured.
2. A leak test is given to assemblies which are hermetically sealed. G. M. counters, ionization chambers, electrometer tubes, etc. are considered self-checking and are exempted from this test.
3. A humidity test is given to prototype assemblies and observatories. The test conditions are 95 percent relative humidity at a temperature of 30°C for 24 hours. Only survival of the exposure is required. Instruments which might be permanently damaged by the exposure may be exempted from the test if precautions are taken to prevent the exposure of those assemblies throughout the program.
4. Vibration tests are given to all assemblies and to the complete observatories. Figure 2 shows the structural design model mounted on a vibration table for one of the transverse axis tests. The vibration tests include both sinusoidal-swept-frequency and random motion tests in the frequency range of 10 to 2000 cps. The expected vibration level for experiment assemblies varies, depending on the location within the observatory, reaching a value of 9.3 g rms at certain frequencies and locations. The expected random motion vibration level is 11.3 g rms with an equal power distribution in the frequency range of 20 to 2000 cps.
5. A linear acceleration test is given to the prototype assemblies and observatories by using a centrifuge. The maximum expected acceleration is 9 g.

Thermal Tests

The experiment thermal tests determine whether the instrumentation operates properly throughout the range of temperatures expected in orbit, whether heat produced within the assemblies can be carried away in the absence of a convective atmosphere, and whether the solar inputs to portions of the detectors exposed to the sun will create excessive thermal gradients. The complete observatory is given thermal tests for these same reasons and, in addition, to check the operation of the active spacecraft thermal control subsystems. Experiment assemblies which are located in the spacecraft's main body are expected to operate throughout the temperature range from -5° to 45°C and those located in appendages must operate over the range from -10° to 55°C.

A test in a thermal-vacuum chamber determines the ability of the experiments to operate without convective heat transfer at temperature extremes for extended periods. Assemblies in which detectors are exposed through thermally significant openings in the spacecraft thermal radiation shield are given solar simulation tests. For this the assemblies are located inside a vacuum chamber, the chamber wall temperature and experiment mounting plate temperature are controlled, and an arc light provides a simulated solar input.

Magnetic Field Tests

The very sensitive magnetometer experiments impose the requirement that the magnetic fields produced by the spacecraft and other experiments be extremely low. Electrical current paths are arranged to produce a very low stray field. Nonmagnetic materials are used wherever possible, but some magnetic materials are necessary for motors, relays, etc. It is required that these magnetic circuits be very efficient to minimize the stray fields. To ensure that these measures have been effective, the magnetic fields of each assembly and the complete observatory are measured. At the present time the permanent, magnetically induced, and electrically induced steady state fields are measured. It may be necessary in the future to add measurements of the ac fields for devices employing power transformers and rotating components.

Integration of Experiments into the Spacecraft and Experiment Calibration

Upon completion of the fabrication and testing of the spacecraft assemblies, they are installed in the spacecraft structure. In general, the integration is done one subsystem at a time, and operational tests are made as each assembly is installed. Test vans have been built to assist in the integration of the OGO observatory and in its checkout throughout the observatory testing program and during the launch operation. The interior of one of these vans is shown in Figure 15. It is capable of supplying numerous controlling signals to the subsystems and displaying data from the subsystems so that the operators can verify that the spacecraft is operating properly.

After the spacecraft subsystems are completely integrated and subsystem interference tests have been made, the integration of the experiments begins. One at a time they are mounted in the spacecraft and electrically connected. Their operation is checked singly through the spacecraft data handling system. They are then operated in groups and, finally, all together, in a check for interference between experiments and between the spacecraft subsystems and experiments.

Following the complete integration of the observatory the experiments are calibrated and the observatory environmental testing is begun. During the various phases of this program, the complete observatory is given an integrated systems test (IST) with the use of the test van. A long series of checks is automatically sequenced and the data from the data handling system are compared with present limits. The test continues automatically as long as the data remain within the predicted limits. If an out-of-tolerance condition occurs, an indication of the nature of the condition is given to the van operator, and the automatic sequence stops until the operator initiates further action. Several thousand checks are necessary to completely verify the condition of the observatory. Checks of the experiments may be included in the automatic test sequence when possible. If it is not possible to make the checkout of some of the experiments semiautomatic, they are checked out manually. Even with the use of the automatic checkout equipment, a complete IST is expected to require several days.

At several points in the observatory environmental test sequence, time is set aside for extensive calibration and checkout of the experiments by the experimenters. An additional period is reserved for this purpose just prior to the shipment of the flight observatories to the launch site.

A shorter period is reserved for calibration at the launch site during the final observatory checkout in the hangar. During these periods, the experimenters will have exclusive access to the observatory for tests of their choice, to give them the best possible chance of knowing the detailed characteristics of their instruments when mounted on the spacecraft.

Launch Site Operations

The observatory will be shipped to the launch site with the appendages, solar panels, and certain critical components removed. Upon arrival in the preparation hangar at the launch site another integrated systems test will be performed to ensure that the observatory was not damaged in transit. The observatory will be completely assembled, aligned, and weighed; its center of gravity will be determined; and it will be mechanically and electrically checked. It will be carried

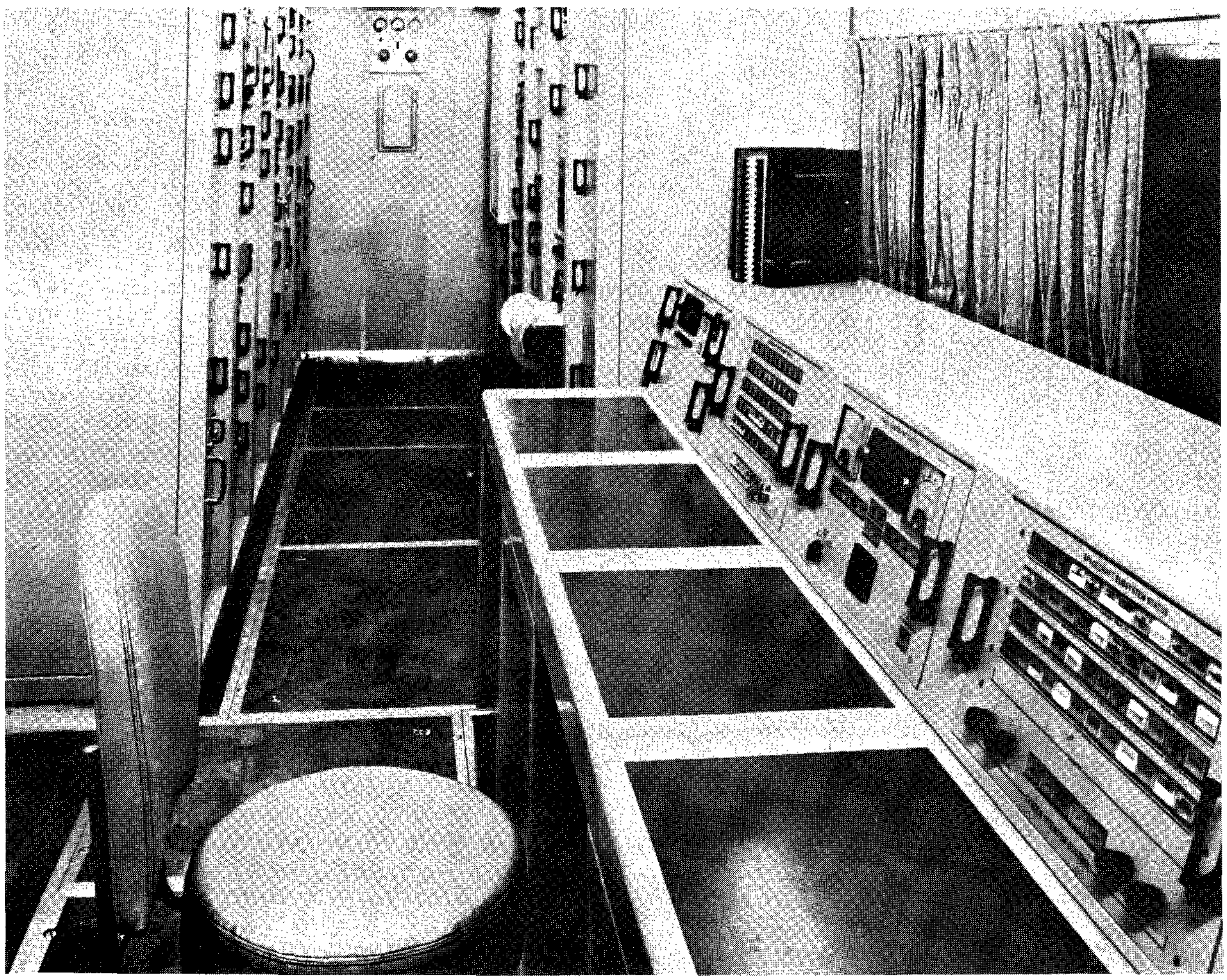

Figure 15—The interior of a test van. The operator's console is at the right and the rest of the equipment is mounted in the racks at the rear. The van contains a complete telemetry and ground command station, and semiautomatic test equipment to perform rapid tests on the spacecraft subsystems. (Photograph courtesy of the Space Technology Laboratories, Inc.).

to the launch pad in its folded configuration, for mating with the launch vehicle. After mating an abbreviated IST will be performed and the shroud will be installed over the observatory. After additional testing the observatory will be ready for launch.

DATA ACQUISITION, TRACKING, AND DATA PROCESSING

An earth satellite of the size of the OGO is capable of producing an extremely large quantity of data. If the first observatory operates properly for the entire planned 1 year period, it will provide approximately 2×10^{10} nine-bit measurements. Since the data are of no value unless they can be analyzed selectively and efficiently by the experimenters, a great deal of attention has been given to the development of a ground support complex which will give each experimenter the data he needs in an easily usable form as rapidly as possible.

This ground station system (Figure 16) includes networks of data acquisition and satellite tracking stations, a control center, a quick-look facility, and a data processing line. The data from the spacecraft are tape recorded at the data acquisition stations, and the tapes are mailed to the OGO Control Center. The World-Wide Satellite and range and range rate tracking stations forward tracking data by teletype to the Control Center where they are used for computation of the orbital parameters. Command instructions are transmitted to the observatory from the data acquisition stations after being generated there or received from the Control Center.

Telemetry data from one of the data acquisition stations, located in Rosman, North Carolina, are relayed directly to the Goddard Space Flight Center by means of a wideband microwave relay link. These real time data are processed at the OGO Control Center for quick-look purposes when it is necessary to react rapidly to conditions on the observatory.

The tape-recorded data from all acquisition stations are processed by the OGO Production Processing Line. At the completion of this processing, individual digital computer tapes are produced for each experimenter, containing the data from his own experiment, timing information, observatory housekeeping parameters (spacecraft temperatures, voltages, etc.), and orbital data. The experimenters will receive two sets of tapes, one containing data telemetered from the observatory and timing information, and the other containing the orbital data and timing information. The merging of these data by the use of the common timing information will be done by the experimenters as part of their more complete data analysis. This procedure will avoid delaying of the telemetry data processing until the final orbit is determined, and will permit easier updating of data tapes if this is necessary.

The production data processing is primarily a sorting operation providing the experimenters with raw data as telemetered from their instruments. All calibrations, corrections, coordinate transformations, etc. will be computed by the experimenters as a part of their data analysis, so that they may maintain close supervision over these operations.

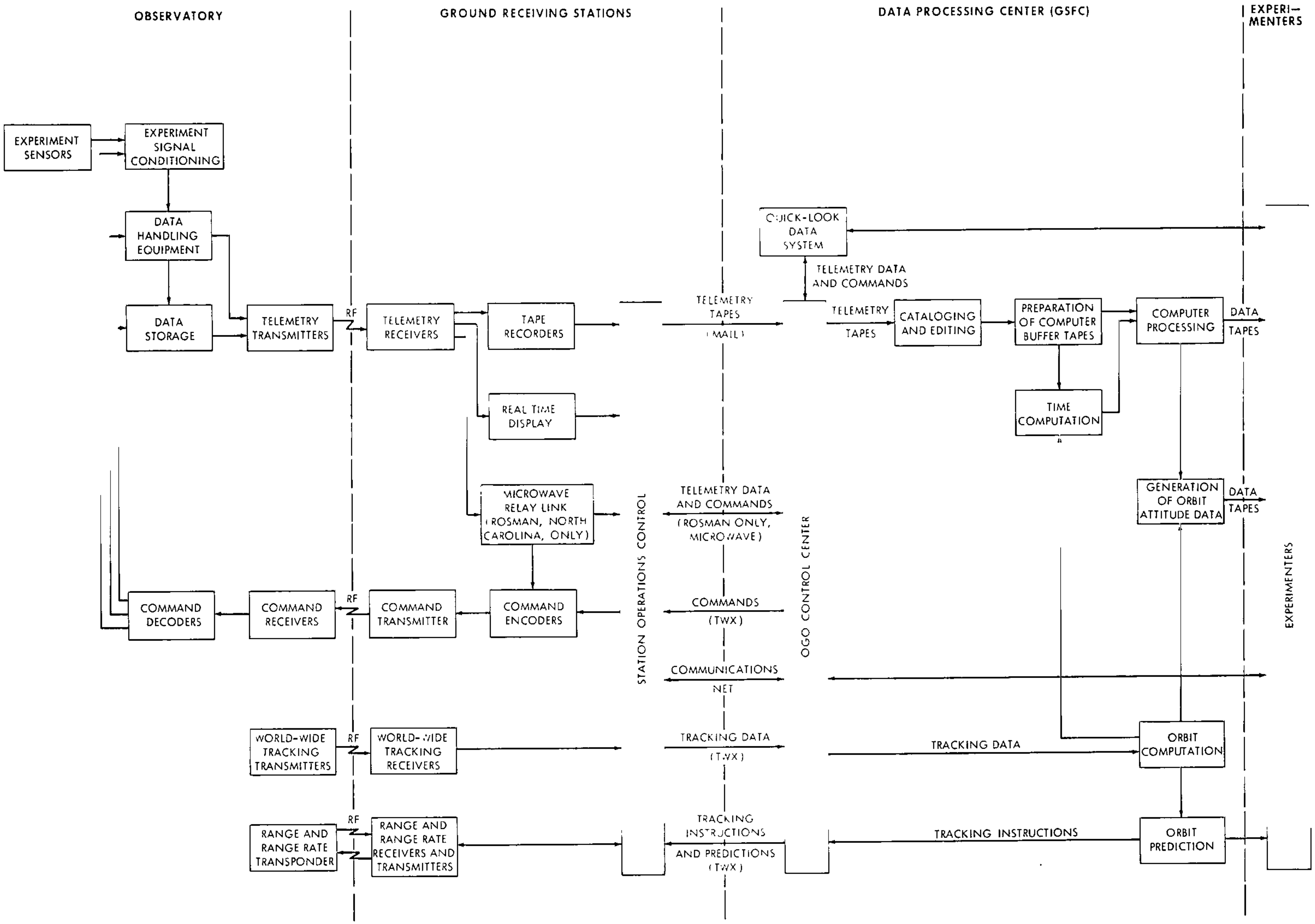

Figure 16—A flow chart for the operation of the OGO in orbit.

Data Acquisition

Data acquisition for both the EGO and POGO will be accomplished by primary and secondary stations with the capability of receiving and recording both the digital and special purpose telemetry at the maximum data rates. The signals will be demodulated and recorded on magnetic tapes, which will be forwarded to Goddard for processing.

The differences between the primary and secondary stations are in the sizes of the receiving antennas and the capability for local data processing. The primary stations have 26 m diameter parabolic antennas, both tone and digital command encoders and transmitters, and PCM data handling equipment to permit limited decommutation and data display for use in controlling the observatory subsystems. The secondary stations have 12 m diameter parabolic antennas, tone command facilities only, and no decommutation equipment. Primary stations are located at Rosman, North Carolina, and Fairbanks, Alaska, and secondary stations are located at Johannesburg, South Africa, and Quito, Ecuador. These four stations will provide EGO orbital coverage about 95 percent of the time, and POGO orbital coverage about 11 percent of the time. Of course, the on-board tape recorders afford 100 percent coverage for digital data at the lower data bit rates.

Tracking

The World-Wide Satellite Network will track the OGO's by the RF interferometer technique. This is an angle measuring system whose accuracy is expected to be marginal for the EGO mission since that observatory spends a large fraction of its time at large distances from the earth. Minitrack stations are located at Antofagasta, Chile, Blossom Point, Maryland, College, Alaska, East Grand Forks, Minnesota, Fort Myers, Florida, Johannesburg, South Africa, Lima, Peru, Goldstone, California, Saint Johns, Newfoundland, Quito, Ecuador, Santiago, Chile, Woomera, Australia, and Winkfield, England. A range and range rate (two-way Doppler) system will also be used to permit a much more accurate and rapid orbit determination. Range and range rate stations are located at Carnarvon, Australia, Johannesburg, South Africa, and Rosman, North Carolina.

Tracking data from both types of station will be sent via teletype to GSFC where they will be used for orbit calculations. The orbital information will be furnished to the experimenters and to all ground receiving stations to permit them to properly direct their antennas toward the observatory.

The Quick-Look Data System

The quick-look data system, consisting of the Rosman primary data acquisition station, the Rosman-Goddard microwave relay link, and the OGO Control Center at Goddard, will permit the display of telemetered data at Goddard on a real time basis, and the immediate transmission of responding commands to the observatory. The microwave relay link has adequate bandwidth to relay the digital and special purpose telemetry to Goddard simultaneously. The return link can relay digital or tone commands.

The arrangement of the OGO Control Center is shown in Figure 17. Incoming digital data are routed to the PCM decommutation equipment which preconditions the signals, detects synchronization, and formats the data for direct entry into the computer. The computer is similar in general to the ones used for checkout of the experiments before launch.

A number of uses of the quick-look data system are planned:

1. Spacecraft status checks—The appropriate program is selected and read into the computer memory. The program causes the computer to select the telemetry words containing spacecraft instrumentation data. Each data word is compared with predefined upper and lower limits in determining if its value is within the range of acceptability. In addition, the data word is converted into its true value (in terms of engineering units) and can be printed out in a form intelligible to station personnel. A printout of all out-of-tolerance data for each spacecraft subsystem can be obtained upon request (by pushing a button on the control and display console) by the station operator. A printout of all the data is available from a 300 per minute line printer. This printout will be kept on file as a permanent record of spacecraft performance.

2. Experiment data processing—The computer can be programmed to perform routine status checks on experiments in much the same manner that they are performed on spacecraft systems.

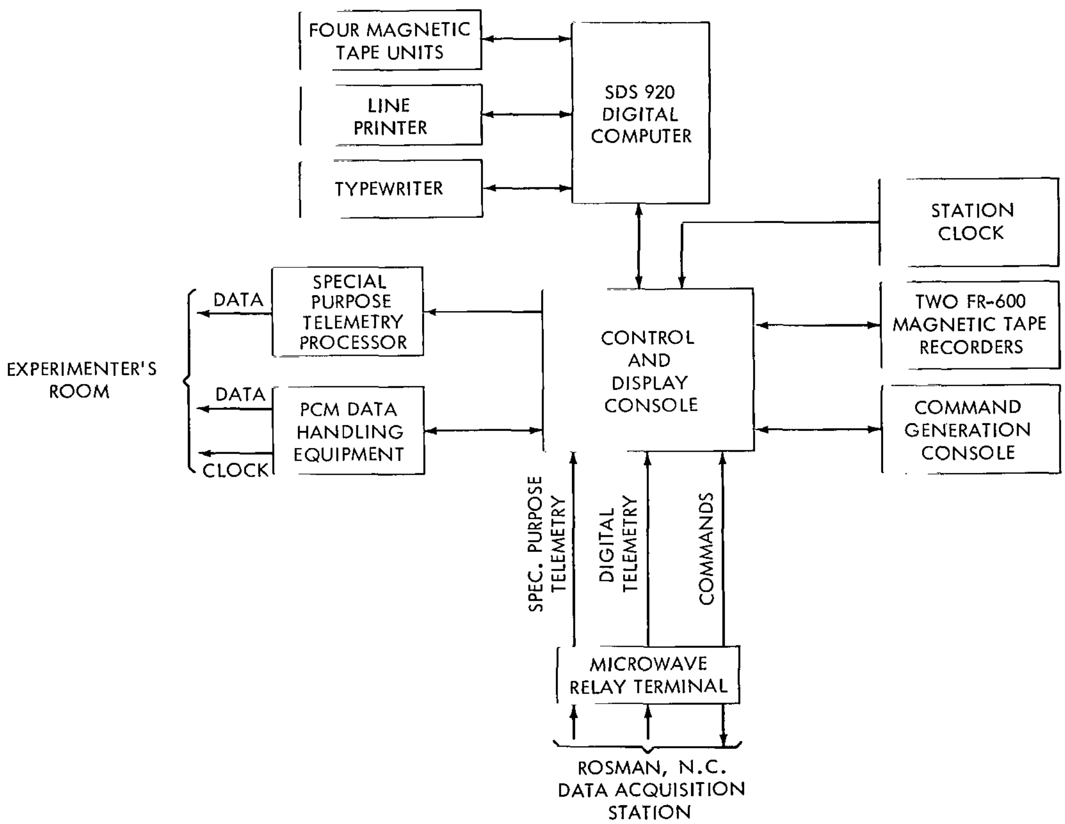

Figure 17—Functional diagram of the OGO Control Center showing details of the quick-look data system.

In addition, programs or sets of programs for each experiment aboard the spacecraft will permit a quick look at the experiment data for evaluating current performance, for calibration, and for monitoring the occurrence of especially significant events when a quick command response may be desired.

3. The experimenter's special data processing equipment—The composite special-purpose telemetry signal will be available in an experimenter's room near the Control Center. An experimenter can locate his own special equipment there to process his data in any manner he may desire.

4. Command initiation—The computer can be programmed to initiate selected commands through the command console in response to specified results of the status checks. Alternatively, commands can be initiated manually. The commands are routed through the microwave link to the command transmitter at the Rosman station and then to the observatory. Commands can also be relayed through the other data acquisition stations by teletype.

5. Data Recording—Two magnetic tape recorders record the incoming telemetry signals, ground time (GMT), and housekeeping data. The recorded data may be played back into the quick-look data system, after a real time pass has been completed, to provide the opportunity for investigating in greater detail the nature of unusual events which may have occurred during the real time pass.

Production Data Processing

The production data processing system converts data from the data acquisition station magnetic recording tapes into a form which is easily used by the experimenters (Figure 18). This processing operation is divided into four phases:

1. Phase I, production of computer-compatible buffer tapes—Depending upon the location of the data acquisition station, shipment of tapes from the station to GSFC may take from 2 days to 2 weeks. Upon receipt, the tapes will be logged and the contents of the tape will be examined to permit a quick feedback to the data acquisition stations for rapid correction in the event of equipment malfunction or incorrect procedures.

The method used in handling the tapes will depend on the bit rate in use and whether the data are real time or on-board recorded, so that the buffer tape created as an end result of the Phase I operation will contain the data in a uniform format. For the first few weeks after launch, tapes will be processed in the order in which they are received; after that, they will be processed chronologically.

The primary functions of the signal conditioner are to establish bit synchronization and to reconstruct the data in a noise-free form. The data processor establishes frame and submultiplexer sequence synchronization. It also combines the data with the acquisition station time.

2. Phase II, generation of the individual experimenter's tapes—This operation will use a digital computer whose ultimate outputs will be demultiplexed experimenter data tapes and an

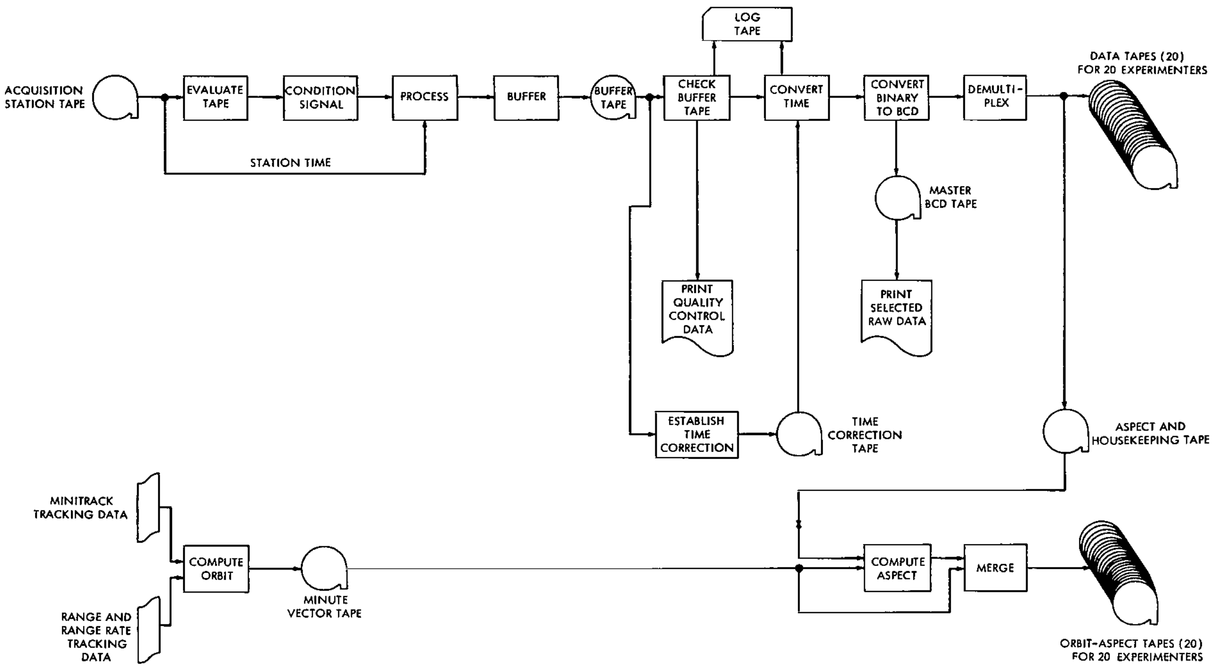

Figure 18—Flow chart for the processing of OGO data and the computation of the orbits.

aspect-housekeeping tape. The computer will also ascertain whether the data quality has deteriorated in the satellite during telemetry or in the Phase I operation.

The task of correlating universal time with the data will be performed during the time-conversion portion of Phase II. A time-correction tape, generated as a part of the Phase III operation, will contain a lookup table listing universal time as a function of the contents of the spacecraft time word register. In order to associate a particular data frame with the proper time, it will be necessary to examine the contents of the telemetered time words, find the corresponding universal time on the time-conversion tape, and insert this value in the time field. This scheme will provide an ultimate accuracy of about 6 msec.

A master BCD tape containing all data will be generated for further use by the processing personnel on an off-line basis. The demultiplexing routine is programmed to place only the information needed by each individual experimenter on his digital data tape.

The processing of the special purpose telemetry is tailored to the detailed requirements of the experimenters who use it. In general, copies of the demodulated signal tapes are furnished to the experimenter who is then responsible for further processing.

3. Phase III, production of time conversion information—The Phase III operation will result in the generation of a time-correction tape which will permit the conversion of spacecraft time to corrected universal time. The spacecraft time contained on the real time telemetry tapes and the acquisition-station tape will be compared on a routine basis. Propagation delays in transmitting

data from the satellite to the stations (about 0.4 sec from the EGO apogee) and from standard time station WWV to the receiving stations (0.058 sec for Woomera, Australia) will be taken into account, as well as any time variations and discontinuities, if they exist, of the basic spacecraft clock. Data time can be determined with an accuracy of about 1 second if the available numbers are used and no additional computations are performed; 6 msec accuracy is possible by performing a timing interpolation.

4. Phase IV, generation of the orbit-attitude tapes—An orbit tape obtained from the standard orbital parameter computation and the aspect housekeeping tape serve as inputs to this phase. The output orbit-attitude tapes contain a block of information for each minute of observatory lifetime. Included on these output tapes are time, satellite position and velocity, the location of the subsatellite point on the earth's surface, the ideal observatory orientation computed on the basis of the orbital information only, and the actual orientation computed with the knowledge of the attitude control subsystem error angles and array angles.

CONCLUSION

The Orbiting Geophysical Observatory program includes the development and use of a large standard spacecraft, the necessary testing and calibration equipment and techniques, a data acquisition and tracking ground station network, and a data processing system. It provides support for the development of experiments and the analysis of the data. Thus, it provides a suitable working environment for the experiments in space, and delivers data to the experimenters in a form suitable for entry into a computer for analysis. The observatories will allow scientists to perform large numbers of geophysical experiments in a variety of possible orbits for extended periods. The scientists retain the full responsibility for the development and preparation of the experiment instrumentation and for the analysis and publication of the data from their experiments. It is hoped that this observatory concept will be of great value in the investigation of phenomena in space.

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